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J. J. Shafer, Committee Chairman

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FOREWORD

Early in April 1960, the Jet Propulsion Laboratory initiated a preliminary study of the utilization of electric propulsion systems in spacecraft. On July 19, members of the study group made interim reports on the status of their work to the Laboratory management. Their presentations, which consisted essentially of orientation and an enunciation of major spacecraft problem areas, are submitted here as an internal report to the Laboratory and for general reference purposes. The original material was modified slightly to answer several questions raised at the presentation and again during the editing of this report when a number of significant comments and recommendations were incorporated because of further study in the intervening period. The Committee is continuing its study.

ABSTRACT

Interim results of a preliminary study of the utilization of electric propulsion systems in spacecraft are presented. Particular attention has been given to the utilization of the *Snap-VIII* reactor and power conversion system as it might be used with an ion motor.

The missions, trajectories, and performance which might be expected from an ion-powered spacecraft are discussed along with the problems in the areas of configuration, guidance and control, and telecommunications which are unique to a spacecraft of this kind.

Recommendations concerning feasibility and plans for the future development of electric propulsion systems are presented in summary.

I. GENERAL ORIENTATION

A. Introduction

J. I. Shafer

In the last few years there has been a strong upsurge of interest in the potentialities, limitations, and characteristics of nuclear and nuclear-electric propulsion systems. One might ask why the Jet Propulsion Laboratory, and others, are suddenly starting to examine these systems as potential supplements to the well-established and proven chemical systems. There are at least three significant reasons:

(1) The nation acquired the ability in January 1958 to impart orbital velocities to a spacecraft; electric propulsion systems, which inherently cannot take off from Earth because of their low thrust, suddenly began to appear feasible when given a chemical boost into orbit.

(2) Advances in nuclear technology indicated that the lightweight source of power for advanced propulsion was potentially available.

(3) Last December NASA Headquarters indicated that JPL would be involved with spacecraft for lunar and planetary missions. Because these are high-energy missions, for which advanced propulsion promises outstanding performance, JPL will obviously be concerned with electric and/or nuclear propulsion for spacecraft.

Table 1 indicates further why many organizations have become interested. It may be noted that the maximum specific impulse that can be derived from chemical propulsion systems is about 500 sec, from direct nuclear fission systems about 1100 sec, and from plasma arc heating about 2000 sec. The magnetohydrodynamic plasma (MHD) and the ion systems are capable of delivering much higher values – up to 25,000 and 60,000 sec respectively; however, values closer to 3000 to 10,000 sec will usually be chosen.

It is of interest also to note that spacecraft thrust-to-weight ratios fall off to unusually low values as the

Table 1. Ranges of specific impulse, thrust-to-weight ratio, and duration for chemical and some advanced propulsion systems

Propulsion Type	Vacuum Specific Impulse sec	Thrust-to-Weight Ratio g	Typical Duration
Chemical	200-500	0.01 to 10	hours to minutes
Direct nuclear fission (heat transfer)	500-1100	0.01 to 10	hours to minutes
Plasma arc heating	400-2000	10^{-2} to 10^{-4}	days
Plasma (MHD)	1500-25,000	10^{-3} to 10^{-5}	weeks
Ion	3000-60,000	10^{-4} to 10^{-6}	months

specific impulse increases. For example, the thrust-to-weight ratio for the ion engine has typical values of 10^{-4} to 10^{-6} g. On the other hand, durations change from hours or minutes for the high-thrust chemical and direct nuclear systems to days, weeks and, for the ion engine, even months of continuous operation.

The specific impulse values shown in Table 1 indicate why interest in the nuclear and electric systems has mounted, but their significance awaits calculation of realistic vehicle payloads; this aspect will be discussed briefly in Sections II-B and II-C of this Report.

Table 2 summarizes the national effort on ion propulsion and plasma propulsion systems. In the ion field there are over nine private companies using their own funds, estimated at one to two million dollars; about three quarters of the funds are for development. The Air Force has let 11 contracts totaling 2.5 million dollars, with about 90% on engine development. The Office of Naval Research and Dr. Stuhlinger of ABMA have six contracts for a total effort of about $670,000, largely development. NASA's Lewis Research Center, active in advanced propulsion work for three years, has approximately 31 professionals working on ion thrust units alone.

In the plasma propulsion field the Air Force has 64 contracts amounting to 3.7 million dollars. Much of this is basic work on plasma control and plasma properties.

NASA's Lewis Research Center has about 19 professionals performing analytical and experimental work on plasma propulsion. In addition, they have approximately 35 professionals performing nuclear research and development. Lewis' facilities include the Plumbrook nuclear reactor and electric propulsion facilities which will be worth seven to eight million dollars by the end of this year. The latter include four high-vacuum tanks 5 ft in diameter and up to 17 ft long, with working pressures of 10^{-6} mm of mercury and liquid nitrogen cooling of the walls.

Table 2. Electric propulsion effort

Company or contractor	Number of contracts	Effort $	Nature
A. Ion propulsion systems			
Private industry	>9	$1,000,000-$2,000,000	70-75% Development
Air Force	11	$2,550,000	90% Development
ONR and ABMA	2 and 4	$ 670,000	75% Development
NASA (Lewis) internal	.....	approx. 31 man-yrs.	Analyses and experiment
B. Plasma propulsion systems			
Private industry	a	a	a
Air Force	64	$3,700,000	60-75% Basic research
ONR and ABMA	2 and 1	$ 450,000	70% Development
NASA (Lewis) internal	.....	approx. 19 man-yrs.	Analyses and experiments

a Not available

Work at the Marshall Center, under Dr. Stuhlinger, consists essentially of system and theoretical analyses; usually they let hardware contracts to industry. Three months ago they had 10 professional people and are still growing.

Two other major contracts of special interest to the Laboratory, because we are considering using the components on *Atlas-Centaur,* are the *Snap-VIII* reactor contract let to Atomics International by the AEC and the *Snap-VIII* power conversion contract let to Aerojet-General by NASA Headquarters. Both contracts were instigated by the Lewis Research Center which will monitor and coordinate for General Ostrander at Headquarters.

The Jet Propulsion Laboratory, too, has had its strong advocates for advanced propulsion work. J. D. Burke, as early as 1951 or 1952, suggested it at frequent intervals. Later, D. Bartz's Section studied nuclear propulsion and its applicability. The Physics Section has, of course, broadened and increased our nuclear efforts.

The electric propulsion studies at the Laboratory, on the other hand, were initiated recently by Dr. J. Lucas and Dwain Spencer. As a result of their efforts, one or more members from each of the Laboratory's technical divisions were invited about six months ago to study integration of electric propulsion systems in spacecraft. In particular, this committee was asked to:

(1) determine whether it was technically feasible to utilize electric propulsion

(2) determine whether today's state of the art was adequate for the engineering task

(3) recommend design criteria for the first electric propulsion system for a deep-space mission

In implementing this charter, the committee quickly decided against a very generalized survey and selected, instead, a specific task. By selecting a specific task, and a mission that was representative of difficult future missions, detailed problem areas would be exposed and evaluated more quickly.

The group elected to study an ion engine based on a *Snap-VIII* reactor system to go aboard an *Atlas-Centaur* vehicle with a Mars satellite mission in the year 1966.

Other missions and vehicles would also be examined, however, when appropriate.

The reasons for selecting these specific quantities are worth noting. Of the many advanced propulsion systems under consideration, the arc plasma and ion engines will probably be developed earliest. Of these two, the arc plasma seems suitable for orbital corrections and orbital transfer missions near the Earth. As planetary and other very high energy missions are considered, however, the ion engine rapidly supplants it in payload capability and is therefore of greater interest to JPL.

The *Atlas-Centaur* vehicle was selected because it probably represents the smallest NASA vehicle which can perform a meaningful mission with an electric-powered spacecraft. In addition the *Centaur* should be relatively reliable before an experimental electric spacecraft is placed on top for a flight. This reliability should be well established by 1966.

The 60 Kw[1] *Snap-VIII* reactor and converter were chosen because calculations by the Lewis Research Center had shown that this power level was roughly the optimum size for the *Atlas-Centaur* vehicle. Furthermore, the long lead time normally required for development of a reactor-power conversion system could be cut drastically by capitalizing on the design and experience of the 3 Kw *Snap-II* program, which had been under development for over 3 years; the *Snap-VIII* system is a scaled-up *Snap-II*. Our own quick survey of the reactor-converter requirements tended to confirm the Lewis studies. However, the fact that the report places major emphasis on the *Snap-VIII* system and the *Centaur* does not necessarily imply that the committee endorses the system. At this early stage in the study the committee is evaluating it, along with alternate systems, when used in a spacecraft for a mission. NASA's contract commitments on *Snap-VIII*, however, suggest a high priority for it.

A Mars satellite appeared more desirable as a mission than a fly-by or a hard landing. Its study would reveal problems that are nonexistent for the simpler missions. In addition, a Mars orbiter seemed to have some performance advantage and might well prove to be a very desirable mission to undertake.

Finally, the flight date of 1966 was set by the long lead time required to develop a power conversion unit which must operate reliably in flight for one full year.

Before examining some system and spacecraft component details, it seems desirable to point out several major differences between the ion and chemical spacecraft for a Mars satellite mission by reviewing the operational procedure. Refer to Table 3 and Fig. 1. In Fig. 1 the spiral paths about Earth and Mars have been enlarged out of proportion to illustrate the nature of the trajectories.

At launch the nuclear reactor and ion engine are not operating. The boost by the *Atlas-Centaur* into a 300-mi orbit resembles that for the chemical spacecraft.

At this point Range Safety imposes a new operational constraint because the ion spacecraft's nuclear reactor must not produce any radioactive contamination of the atmosphere. Not until the spacecraft has made sufficient revolutions for Range Safety to establish it as definitely in orbit will the ion propulsion system be activated by starting up the reactor and power conversion system.

[1]Kilowatts of electric power

Table 3. Major differences between chemical and ion spacecraft for Mars satellite mission

Operation or characteristic	Chemical spacecraft on *ATLAS-CENTAUR*	Ion spacecraft on *ATLAS-CENTAUR*
Range safety	Normal fires and aborts	Atmospheric contamination from nuclear reactor
Spacecraft acceleration	Midcourse $\cong$ 0.2*g*	Powered flight at 5×10^{-5} g
Earth escape	Short parking orbit	Long spiral orbit under power
Transit time (Earth orbit into Mars orbit)	240 days (typical)	360 days
Telecommunication antenna	— — —	Cannot look through ion beam
Communications	Power limited	Power relatively unlimited
Guidance	— — —	Minor difference because of low acceleration and long time
Midcourse maneuver	Rotate spacecraft, perform correction, rotate and reacquire Sun	Continuous spacecraft maneuver thruout heliocentric orbit into terminal phase; no midcourse rocket
Mars orbit capture	Retro rocket and guidance maneuver	No retro-rocket; deceleration thruout Mars reverse spiral

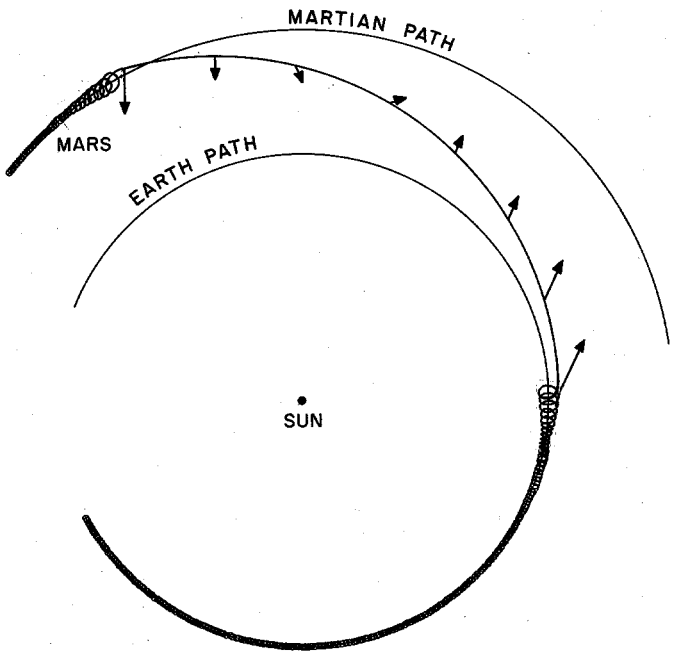


Figure 1. Schematic trajectory for Mars satellite mission

The resultant 0.4-lb thrust should produce a spacecraft acceleration of 5×10^{-5} *g*. This may be compared with a midcourse acceleration for the chemical spacecraft of about 0.2 *g*.

In order to build up to hyperbolic escape velocity using this extremely low acceleration, the ion spacecraft must orbit many revolutions as it spirals out from the Earth (see Fig. 1). Near Mars it must decelerate during a gradual spiral of many revolutions before settling into orbit.

In a Mars satellite mission the ion propulsion system operates continuously from orbit to orbit. Transit time is about 360 days compared to a typical 240 days for the chemical system. When larger ion spacecraft with more power than 60 Kw become available, they will exhibit about the same transit time as the chemical system for this mission; the *Snap-VIII–Centaur* is strongly power limited.

During the heliocentric portion of the trajectory the propulsion thrust vector, represented in Fig. 1 by the arrows, is continually changing, both in magnitude and direction. In general, thrust is directed outward during the first half of the trajectory and inward during the last half.

The telecommunication antenna must be carefully positioned on the spacecraft in order for it to see the Earth at all times. It must not be forced to scan through the plasma beam of the ion engine during spacecraft rotation about the Earth and Mars nor during the extreme spacecraft yawing in the heliocentric orbit; otherwise contact with Earth would be lost.

The communications designers should derive one new and significant advantage with the ion spacecraft, however. Electric power is relatively unlimited during the flight and especially after the spacecraft arrives in orbit. In fact, the availability of 60 Kw of electric power poses quite a challenge to the ingenuity of the communication, guidance, and payload engineers when they are asked to utilize it to its fullest potential.

Guidance components for the ion spacecraft will probably have minor differences from the celestial guidance that is currently being examined for ultimate use on the chemical *Centaur* because of the low acceleration and long-operating time of the former. Section III-C will expand on this aspect somewhat.

The midcourse maneuver in the chemical spacecraft involves, in a relatively short interval of time, rotating the spacecraft, performing an impulsive correction with a separate rocket, then rotating and reacquiring the Sun as reference. In contrast, the ion spacecraft performs a slow and continuous[2] maneuver throughout the entire heliocentric orbit; no separate midcourse rocket is needed.

The Mars orbit capture, in the case of the chemical spacecraft, is accomplished by destroying the excess velocity with a chemical retro-rocket, and a terminal guidance maneuver. The ion spacecraft requires no separate retro-rocket. Instead, deceleration by the ion propulsion system is used throughout the Mars reverse spiral into orbit.

In passing it must be noted that the long-operating life of the ion spacecraft introduces severe problems of reliability. This long operating life implies long lead times in developing components. In addition, during the long flight, damage to operating components by meteoroids must be anticipated and prevented if the mission is to be successful. In fact, this overall problem of reliability probably constitutes the greatest single challenge in the utilization of ion propulsion systems for space missions.

[2]An Irving and Blum optimum program is used here for illustrative purposes. Although attitude control must be capable of functioning over the entire trajectory in order to execute the thrust program, the controls will probably operate as a step function rather than continuously.

B. Basic Spacecraft Configuration

H. Bank

The configuration study of an ion-propelled spacecraft has emphasized the mechanical engineering aspects and was initiated in order to become acquainted with some of the critical problems of integrating ion propulsion with other spacecraft subsystems.

This study started with a review of the major components of the new propulsion system to get an idea of their physical dimensions and to review their operating temperatures and nuclear radiation characteristics. These two conditions, along with large radiator requirements, are probably the most important items influencing the spacecraft configuration. Results of the review are summarized in Table 4.

The nuclear reactor, which operates at approximately 1300°F, introduces a strong radiation field of neutrons and gamma rays. Since this radiation is three or four orders of magnitude above the allowable dose for ordinary electronic components, for a one-year exposure period, it is necessary to provide a relatively large and heavy radiation shield. Note, however, that in the process of absorbing neutrons the temperature of the shield will rise and may approach 800°F.

The power conversion unit has operating temperatures that range from 1300°F for the boiler to 600°F for the large radiators and condensed mercury or working fluid. The radiator is one of the largest spacecraft components; at the 600 to 700°F radiator temperature, approximately *1600 sq ft* of radiation area is required to dissipate the heat of condensation into space. The power conversion unit itself is relatively insensitive to nuclear damage, with the possible exception of some of the electronic controls which can be located beyond the radiation shield.

The 4-ft spherical propellant tank will contain approximately 4000 lb of propellant (liquid cesium, M.P. = 83.3°F). An operating temperature of 400°F will be maintained in order to insure sufficient vapor pressure for pumping the propellant. Temperature regulation will be complicated by propellant heating due to neutron absorption. The cesium itself is considered insensitive to nuclear damage. The ion thrust unit will operate at the highest spacecraft component temperature, 2000°F, and is also relatively insensitive to nuclear damage.

The new and hostile environment of the propulsion components not only affects their design and operation but it also has strong effects on other spacecraft subsystems such as guidance, controls, communication and payload. The most sensitive components within these subsystems are the electronics. Their general temperature limit is in the 100 to 300°F range, although some components do operate at higher temperatures. Those elements which tend to be sensitive to high temperatures are the organic dielectrics which decompose, and the semi-conductors which break down. As it turns out, these same two elements are also sensitive to nuclear damage.

Table 4. Ion propulsion component characteristics (Estimated)

NO.	COMPONENT	SIZE	WEIGHT	OPERATING TEMPERATURE	NUCLEAR RADIATION
1	Reactor	Cylinder 2-ft dia 1½-ft thick	300 lb	1300°F	Emits 10^{18}N/cm/yr of 1 Mev Range
2	Neutron shield	Cylinder 2½-ft or more dia Approx 2½-ft thick	400 lb (or more)	may be above 800°F	Absorbs 10^{4}N/cm/yr of 1 Mev Range
3	Boiler, turbine-generator & pumping equipment	2 Cylinders 1½-ft dia 3-ft thick	1000 lb	1300°F to 700°F	Generally unaffected
4	Radiator	2 Rectangular Sheets 15 x 30 ft ½-in. thick	600 lb	600°F to 700°F	Generally unaffected
5	Propellant tank	4 ft sphere	4300 lb	400°F (desired)	Absorbs neutrons
6	Thrust unit	Cylinder 4½-ft dia ½-ft thick	300 lb	2000°F	Generally insensitive

If ceramic materials were to be used instead of organics, and vacuum tubes were to replace transistors, higher temperature electronic systems might be possible and, even better, the radiation shield might be unnecessary. This would pose quite a challenge to the electronic engineers; because they already have a major task in developing one-year reliability in their equipment, current electronic technology was assumed for this study.

There are, of course, a number of other important problems associated with new propulsion systems which affect the configuration. Some of these are:

(1) The communication antenna problem; its look angle is complicated by the unique yawing of the spacecraft during heliocentric flight. This was discussed briefly in the preceding section.

(2) The look angle requirements of payload optical experiments, and celestial guidance sensors, which also are complicated by the same spacecraft yawing.

Most of these are believed to be of secondary importance to the basic vehicle configuration and will be discussed later.

Having established the general operating characteristics of the subsystems and their general sensitivities to the new temperature and radiation environment, it seemed appropriate to investigate a number of possible vehicle configurations within the *Centaur* booster constraints. It was very evident that the critical configuration problem essentially reduced to:

(1) Establishing acceptable packaging for an efficient radiator

(2) Locating the various subsystems appropriately relative to their temperature and nuclear radiation sensitivity

The radiator problem is a difficult one which many organizations are studying relative to the *Snap-VIII* program. Some of the radiator designs suggested are shown in Fig. 2. The cylindrical configuration, which is similar to the *Snap-II* design, is convenient for a 30-Kw size conversion system but requires articulation for the 60-Kw size. The latter is, of course, the size required for a Mars mission. Only one side radiates heat to space in this configuration, thus doubling the area requirements relative to other designs. However, the radiator can also be used for airframe structure, which tends to offset its weight disadvantage. This design also accentuates the heating problem if the equipment is located internally.

A fan configuration is also under consideration. This design has a fold-out arrangement similar to the ranger solar panels. It involves many rotating parts and a difficult manifolding problem, and doesn't appear very attractive.

The third or wrap-around configuration is designed to be sufficiently flexible in the transverse direction so as to deform elastically around the vehicle framework. It would be unwrapped (See Fig. 2) following boost and just before ion propulsion begins. This configuration was proposed by Aerojet and chosen by NASA for development in the current *Snap-VIII* program. It seems to be the most desirable of the radiator designs and has the advantages of: good packaging, light weight (since both sides radiate), no hinged joints, and no enclosure of other subsystems in its highly thermal environment. However, there is serious concern about the use of the large floppy radiator which could introduce very severe dynamics problems. Natural frequencies of these cantilever structures would be of the order of 0.01 to 0.05 cps which may involve electro-mechanical coupling problems with controls systems. This problem, hopefully, will be investigated soon.

If one considers constraints on the wrap-around radiator, and the size of the other subsystems components, it is evident that a stacking of these components within the radiator cylinder is appropriate. This stacked design has been considered in two arrangements: (a) that which emphasizes temperature sensitivity and (b) that which emphasizes nuclear isolation.

A scale model of a proposed *Snap-VIII* spacecraft best illustrates these two arrangements. Figure 3 shows the model during the vehicle boost phase. The spacecraft would rest on top of the *Centaur* and would be contained within the old *Vega* shroud. Figure 4 shows the same spacecraft with both 15 x 30-ft radiator sections unfolded and as it will appear as the ion engine is activated.

The spacecraft arrangement in Figure 5 locates the major components and subsystems in the order that is best from the standpoint of temperature sensitivity; highest temperature is at the aft end. The 2000°F thrust unit is at the stern next to the 1300°F nuclear reactor. The latter is topped by the 1300 to 600°F power conversion unit and the 400°F propellant tank. These are flanked by the 600 to 700°F radiators. The neutron shield is positioned at the front to protect the 100 to 300°F guidance controls and payload equipment. The antenna is located between radiators on the side in order

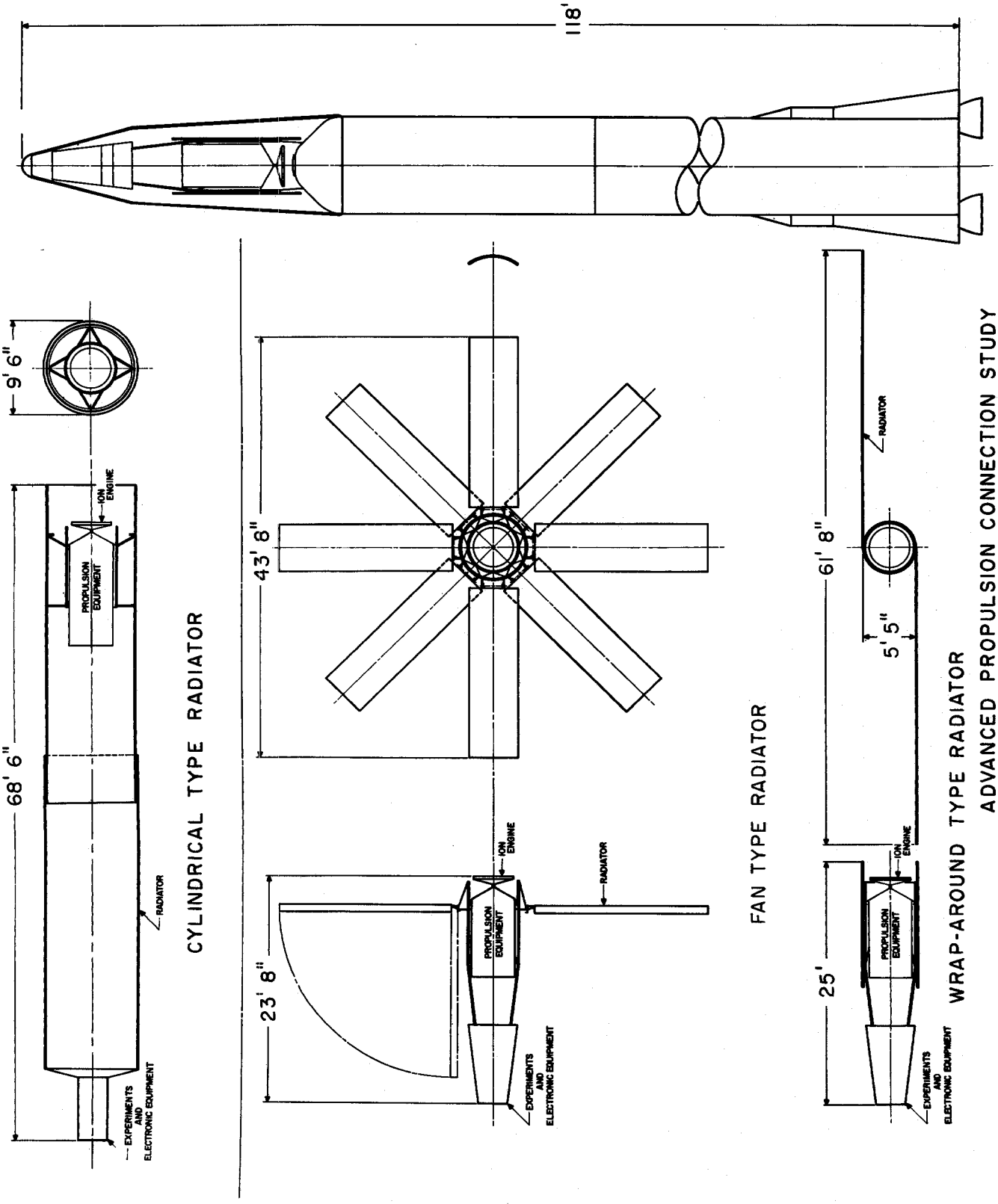


Figure 2. Radiator configurations

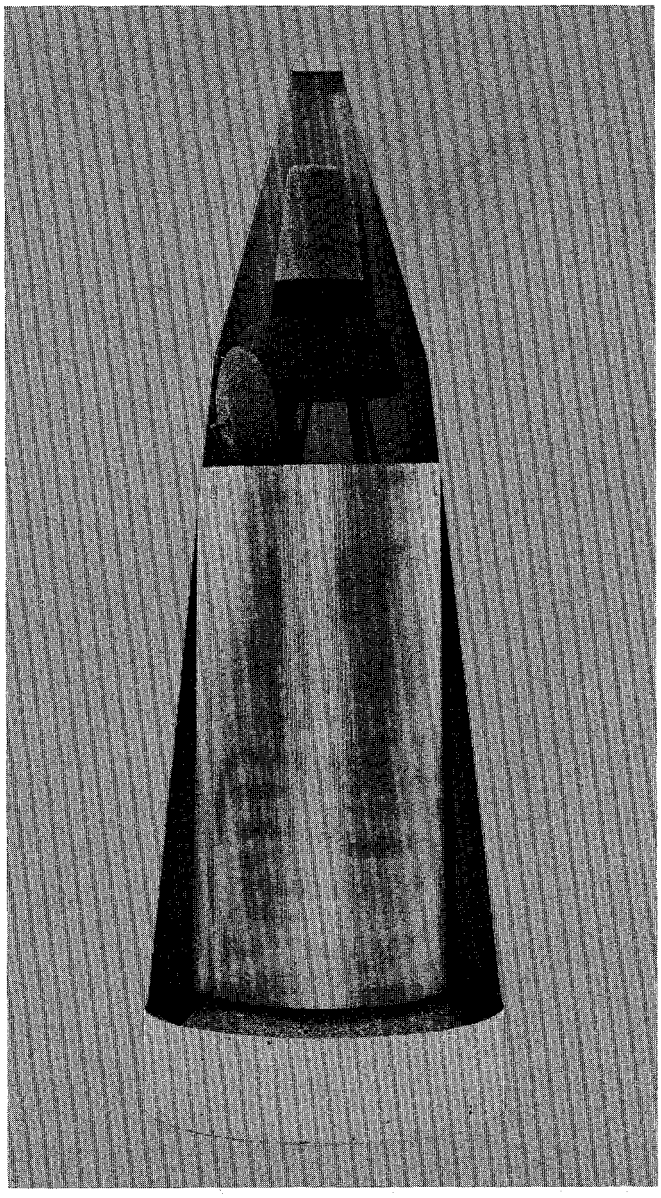


Figure 3. Model of *Snap-VIII* spacecraft during boost phase

to provide a 360 deg angle of view in maintaining contact with the Earth. As shown, it is located near the front of the spacecraft; however, a more detailed review is necessary to determine whether this position is optimum.

Some of the advantages of this temperature-sensitivity arrangement compared to the other configuration, are:

(1) Reduced heating problems, particularly of guidance and controls and payload

(2) Cool reactor shield, since reactor distance and propellant tank provide shielding

(3) Reduced center of gravity travel for guidance and controls

(4) Clean interface between subsystems

(5) Short hot-lines between reactor and conversion system

Some of the disadvantages are:

(1) Look angles for guidance and control and payload sensors may be poor at the forward end

(2) Shield may be relatively large and heavy to protect guidance and control plus payload (depend-in on packaging efficiency)

(3) Active cooling may be required on the propellant tank since it is exposed to the nuclear radiation and has a fairly good absorption cross section

(4) Tighter center-of-gravity alignment is required

(5) Longer fuel lines are required

(6) Problems with booster (*Centaur*) shielding.

The arrangement emphasizing nuclear radiation isolation, as shown in Fig. 6, merely shifts the reactor to the front of the spacecraft and locates the guidance controls and payload either behind the shield or packages it within the aft structure consistent with the thermal, nuclear radiation, and look angle constraints.

Advantages of this design are:

(1) Look angle more flexible for guidance, control, and payload

(2) Possibly smaller and lighter-weight shield

(3) Cooler propellant tank

Disadvantages are:

(1) More heating problems on guidance, control, and payload, and shielding may require considerable active cooling

(2) Poorer interface between subsystems

(3) Long, hot–lines between reactor and conversion system

These two arrangements are being studied in detail to determine the most desirable configuration.

The mechanical configuration is only one of the many vehicle problems requiring study. In addition, the start-up and separation problem is worth noting. If one considers the events after entering the 300-mi parking orbit, it would seem desirable to use the *Centaur* second stage for maintaining attitude control while Range Safety

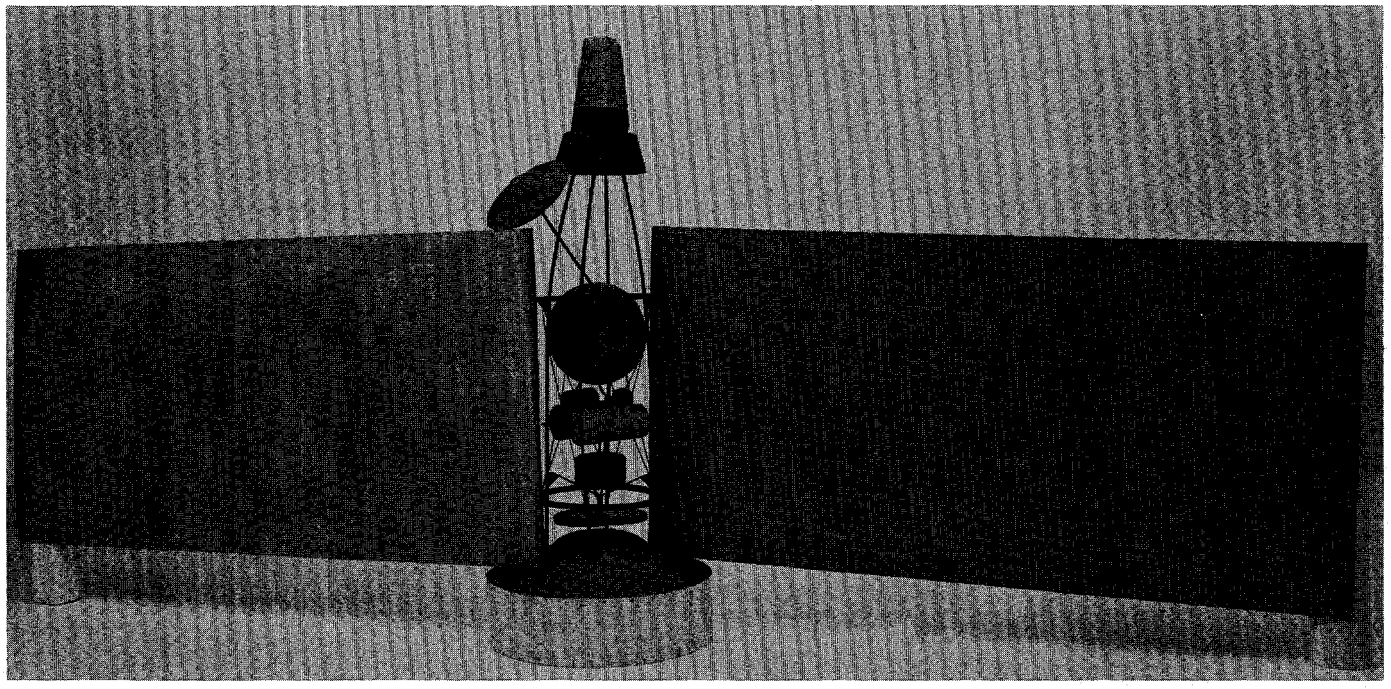


Figure 4. Spacecraft configuration at injection

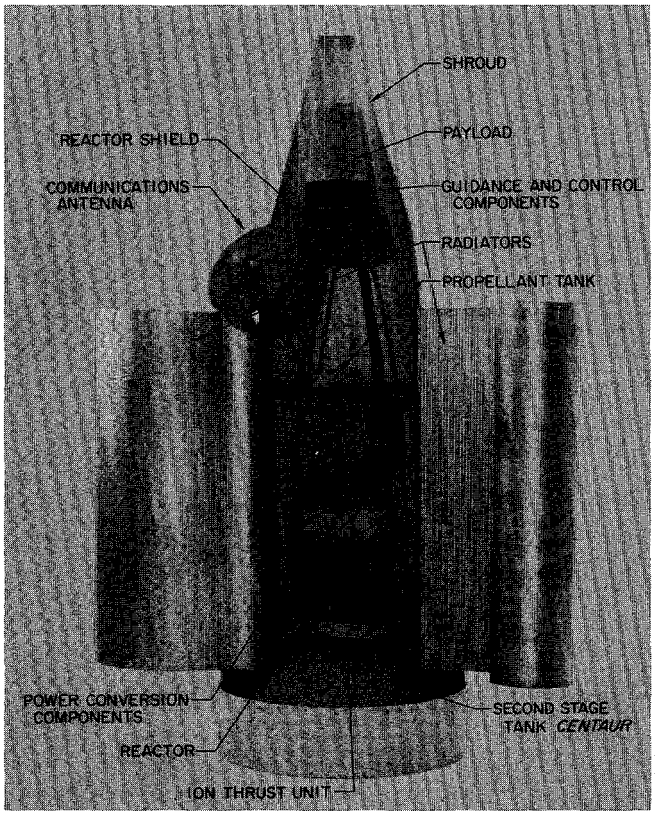


Figure 5. Spacecraft configuration for optimum temperature arrangement

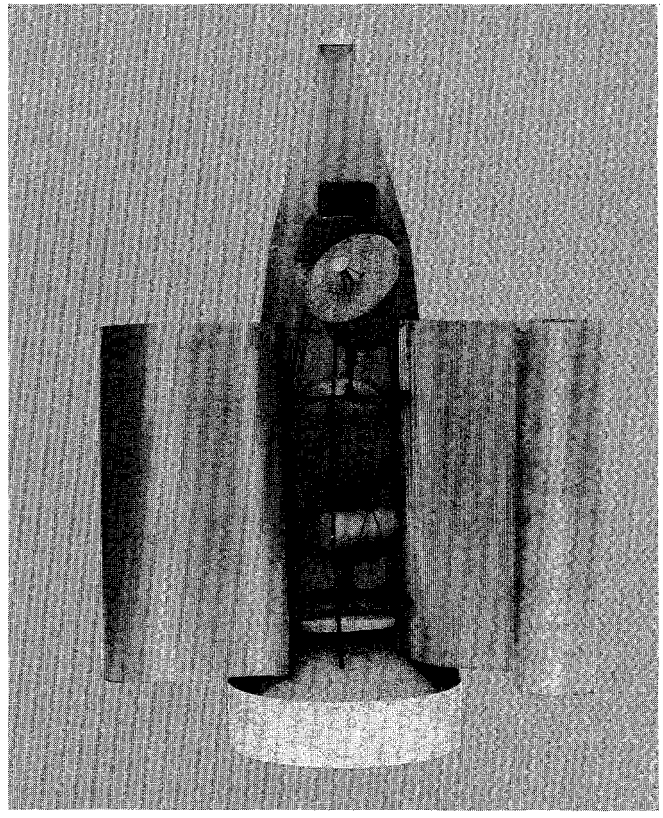


Figure 6. Spacecraft configuration for optimum nuclear radiation arrangement

establishes orbit security for start-up. Upon approval, the proposed sequence would consist of (1) unfolding the radiator, (2) activating the reactor and power converter, and (3) allowing the electric power system to reach equilibrium conditions. The transient electric power could be dissipated in the parasitic load. During this entire period the second stage could be used to take out the undesirable start-up transients. Once the equilibrium condition is established, then stage separation will take place concurrently with the activation of the ion-thrust unit.

There are, of course, many other problem areas such as that of attitude control and the airframe and structural materials problem. These problems arise out of the high temperature and long operating life requirements. Fortunately, in the latter case, considerable technology exists on materials and structures operating at the expected temperatures and it is believed that no new research will be necessary in this field. These, and many other problems, should be solved, at least in concept, prior to the completion of the study in order to establish vehicle engineering feasibility.

II. MISSIONS, TRAJECTORIES, AND PERFORMANCE

A. Scientific Missions For Ion-Powered Spacecraft

M. Neugebauer

There are four missions under consideration by the committee as possible early applications for an ion-powered system. These missions are a Mars satellite, a solar probe, a Jupiter probe, and an out-of-the-ecliptic probe.

The advantage of an advanced-propulsion Mars satellite over an ordinary Mars satellite, from the scientific observations point of view, is that it is possible to spiral slowly into the planet thereby providing an opportunity to obtain complete and accurate values for the radiation belt intensity (if one exists) and for the composition and structure of the martian atmosphere.

The objectives of a solar probe are fairly obvious. The particles and fields measurements on *Pioneer-V*, *Rangers-1* and *2*, and *Atlas–Able V* are aimed at determining the properties of the solar corona in order to learn more about (1) the processes of particle acceleration at the Sun and the trajectories of these particles in interplanetary space and (2) the modulation of galactic cosmic rays by interplanetary fields which may be of solar origin. These measurements must be made close to the Sun as well as very far from it before these phenomena are completely understood. The temperature problems are doubly severe for this mission, so it will probably not be the first ion-powered mission.

A Jupiter probe offers some very exciting possibilities. Jupiter is the closest of the major planets, i.e., those planets which differ radically from the Earth, Mars, Venus, and Mercury, both in composition and in structure. It is important to know more about Jupiter and the other major planets in order to choose between the various theories for the origin and history of the solar system.

Only three or four of the specific experiments which might be carried out by a Jupiter probe will be mentioned at this time. First, it would be of interest to take photographs of some of the satellites of Jupiter. The object in doing this would be to see if the moons of Jupiter look like our Moon, i.e., do they have craters and do they have maria? In this way it would be easier to interpret our Moon's origin and history, for it would then be known whether our Moon was a common occurrence in the solar system or if it were some kind of freak.

Measurements of radio waves from Jupiter have aroused a lot of interest of late. This radiation is of a decidedly non-thermal character and may be due to synchrotron radiation from electrons trapped in Jupiter's magnetic field. Calculations of the magnetic field strength needed to trap such electrons indicate that Jupiter's field must be of the order of kilogauss. It is a real problem to

explain how a planet made out of hydrogen and helium in large part, as we think Jupiter is, can maintain a field of this strength. In order to resolve this problem a magnetometer should be included on an early Jupiter probe to see if the field really is this strong.

Another interesting problem in connection with Jupiter concerns its atmosphere. Because the atmosphere of Jupiter is supposed to be very much like the Earth's atmosphere billions of years ago when life was forming on the Earth, it would be very interesting to look to see if the first steps toward life had taken place on Jupiter. One might do this by examining the absorption spectrum of sunlight reflected from Jupiter for indications of the presence of organic molecules.

Particle and field measurements should be made on the way to Jupiter. Recent theories suggest that the atmosphere and the magnetic field of the Sun extend out to a region somewhere between the Earth and Jupiter. Beyond this there is some sort of a boundary layer whose properties are interesting and unknown. Still farther out there should be a galactic magnetic field and interstellar particles; it would be of interest to know more about them.

The fourth mission for an advanced propulsion system would be an out-of-the-ecliptic space probe. The scientific objectives of such a space probe are many. First, an out-of-the-ecliptic space probe could give data to help answer several cosmological questions, such as: what is the rate of mass loss of the Sun and other stars like it? This net mass flow can only be determined if the rate of flow of matter away from the Sun is known as a function of solar latitude. It is important to know whether or not the Sun and similar stars emit enough matter to account for the observed cosmic radiation.

Another question of cosmological interest that could perhaps be answered by an out-of-the-ecliptic probe is whether or not the observed accumulation of gas and dust near the ecliptic is a fossil accumulation or whether it is the result of a steady-state dynamic process.

A lot of interesting solar physics could be done with an out-of-the-ecliptic probe. The source of low-energy charged particles which are almost continuously emitted from the Sun should be determined; it is not known whether this is an overall evaporation or whether the particles are emitted from active regions on the Sun near sun spots only. The active regions are generally confined to definite latitudes on the Sun, so there may be some sort of latitude effect in low-energy particle radiation. The interaction of this ejected matter with the interplanetary magnetic field at high solar latitudes should also be studied. In order to understand the acceleration mechanism of these particles, it is necessary to know how much energy was expended and how much material was ejected. It is not possible to calculate this unless the angular spread of the ejected material, both in the ecliptic and perpendicular to it, are known.

It would be extremely valuable to have out-of-the-ecliptic space probes both at the solar maximum and solar minimum. Figure 7 shows a predicted sun spot cycle for the next few years. The next minimum will probably be in 1965 or 1966; 1968 will probably be near the next solar maximum. The time of year and the direction of firing should be planned to take advantage of the little over 7-deg inclination of the ecliptic to the solar equator, as the main objective is to get to high solar latitudes.

Figure 8 shows the solar latitudes at which sun spots or active regions appear as a function of time. Just past solar minimum there are few sun spots and active regions, and these spots are at relatively high solar latitudes, say 25 deg. As an example, assume a space probe at 22-deg solar latitude; the Earth at this time is at 7-deg solar latitude. If an active region appears at 25 deg N, there are observing stations (the Earth and the probe) at 3 and 18 deg from the region. On the other hand, if an active region appeared at 25 deg S, there are observing stations at 32 and 47 deg. Thus, as the spots appear in the two hemispheres, only one or very few at a time, the angular dependence of material ejected from active regions near the sun spot can be monitored with four data points: 3, 18, 32, and 47 deg. At solar maximum, however, there are so many sun spots that there is no possibility of following material ejected from just one at a time. Thus it would be nice to get completely beyond the sun-spot zones at this time to 30- or 40-deg solar latitude, which means 23 to 33 deg from the ecliptic, to observe the maximum difference between disturbed and quiet solar regions.

In summary, one shot should be made in 1965 or 1966 and go to solar latitudes of 15 to 25 deg, which means 8 to 18 deg out of the ecliptic. Another probe in 1968 should go to as high a solar latitude as possible; 23 to 33 deg out of the ecliptic would result in a very valuable scientific experiment. Of course, these angular distances are just lower limits; an angle of 90 deg would be more than acceptable.

These out-of-the-ecliptic probes would, of course, carry a whole array of radiation detectors for both high and low-energy particles, as well as a magnetometer of

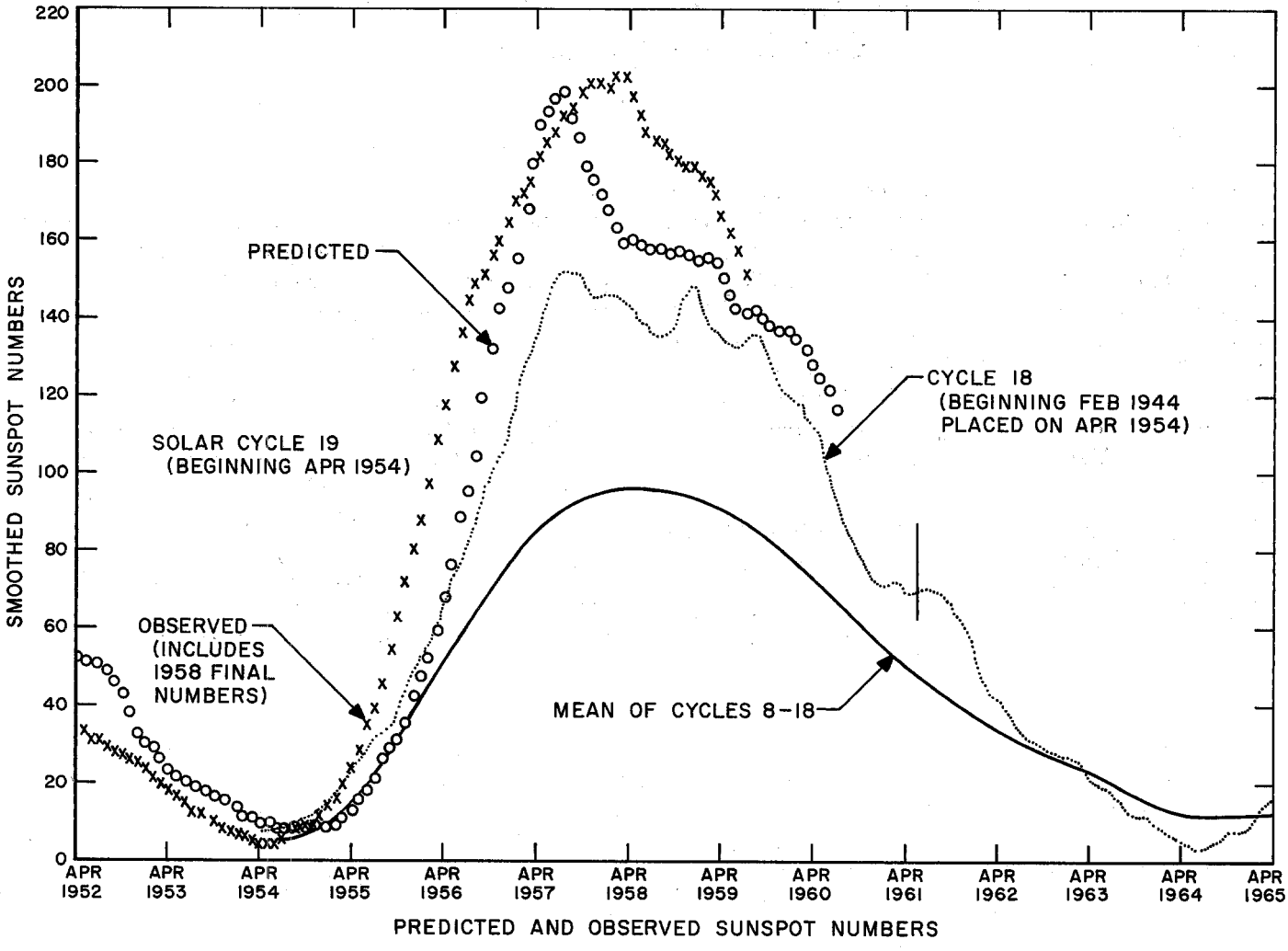


Figure 7. Predicted and observed sunspot numbers

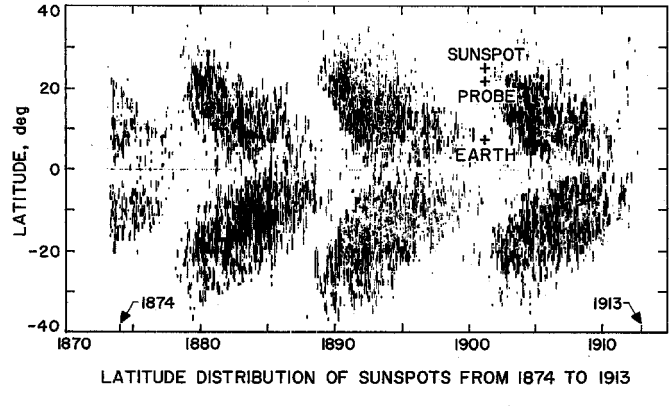


Figure 8. Latitude distribution of sunspots from 1874 to 1913 (Maunder 1922)

some type. Micrometeorite detectors, such as the photomultipliers and microphones on *Rangers-1* and 2, should also be included. Probes which get 25 deg or more from the ecliptic are well above most of the dust which is concentrated in the plane of the ecliptic. A very nice experiment has been suggested by Dr. R. B. Leighton at Caltech which involves taking pictures of the ecliptic from above. These pictures would show whether the dust distribution is fairly uniform or whether it has arms like a spiral galaxy. This is probably the only way of determining whether a pattern exists. Furthermore, if a spectrum of the light reflected from near the ecliptic is obtained, then the electron and dust contributions from this reflected light can be calculated, and distribution of dust and electrons can be found separately.

B. Trajectory and Performance For An Out-of-the-Ecliptic Mission

S. Benesch

Some of the significant observations associated with an out-of-the-ecliptic probe have been presented in Section II-A. Figure 9 shows an isometric drawing of a trajectory for such a probe. The vehicle, a *Centaur* with a *Snap-VIII* electric propulsion system, has a gross weight of 8800 lb, a specific impulse of 3000 sec[3], a propellant weight loss of 12.67 lb/day, and departs from a 300-mi circular coasting orbit about the Earth. Table 5 gives a preliminary weight breakdown for the *Centaur–Snap-VIII* system for (1) a Mars-orbit mission and (2) an out-of-the-ecliptic mission.

The first, or spiral portion of the trajectory, Phase I, was computed on a two-dimensional, two-body program until escape energy was achieved. The results of Phase I were then used as inputs into Phase II of the trajectory, which is a three-dimensional, *N*-body program. The initial point in Phase II was chosen as 0.4 radians prior to the escape energy point on Phase I. Conditions were satisfied such that the asymptote of the escape hyperbolas was aligned perpendicular to the orbit plane resulting when the vehicle's distance from the Earth reached 10^6 Km, at which time the coordinate system was transferred from the Earth to the Sun. The thrust vector is then maintained normal to the orbit plane (i.e., normal to both the radius vector and the velocity vector).

The initial point of the trajectory in Fig. 9 is taken at 155 days after departure from the Earth-centered 300-mi orbit. This time is a few days after escape.

[3]Approximately optimum thrust and specific impulse for the *Centaur* with a *Snap-VIII* electric system (cf., Section III-A-4).

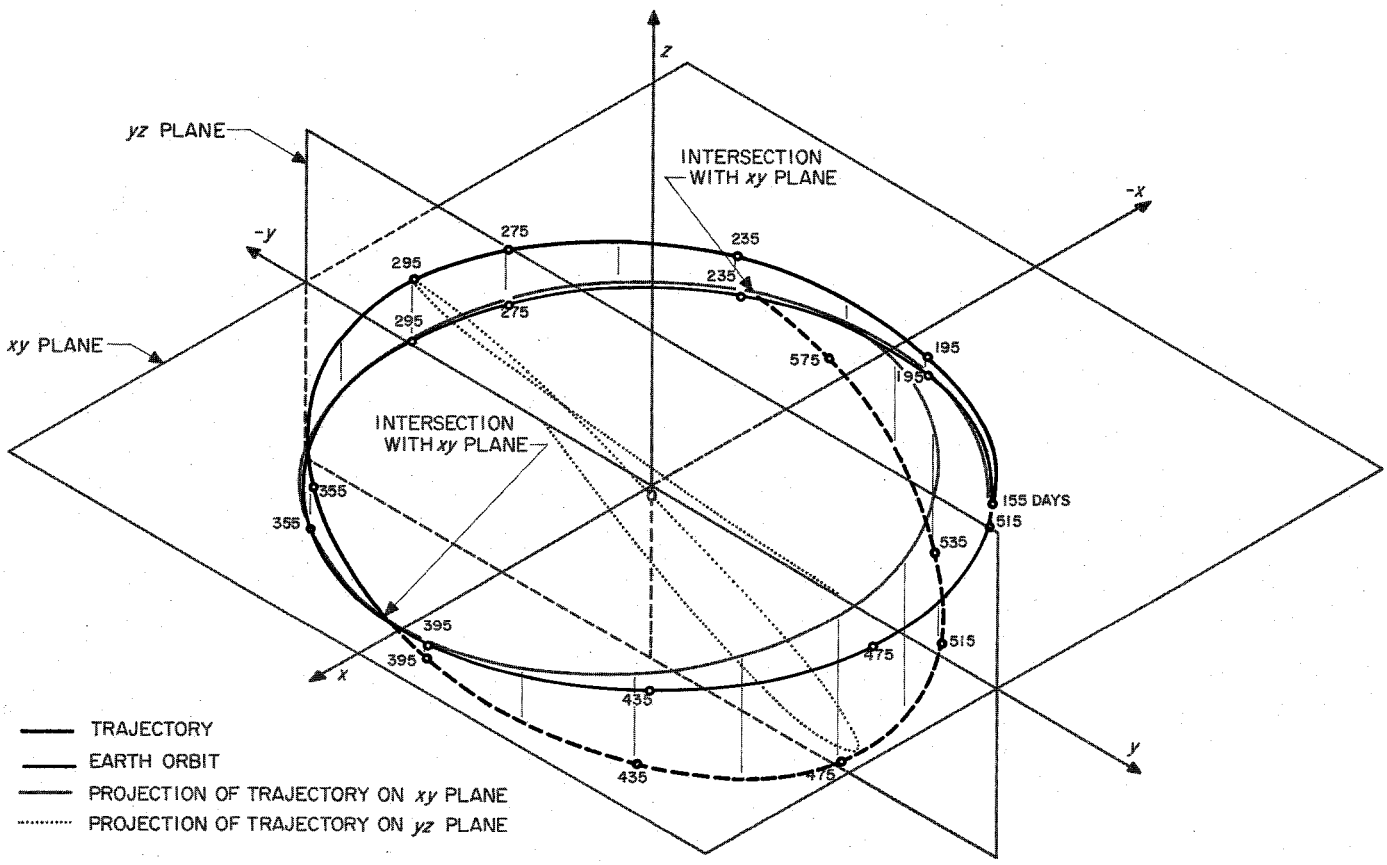


Figure 9. Out-of-ecliptic low-thrust trajectory

Table 5. Preliminary weight breakdown for *Centaur–Snap-VIII* system (Ion stage — 60 kw)

		lb	
Reactor[a]		300	
Shield[a]		500	
Conversion System* (Less Radiator)		1000	
Radiator*		600	
Engine		300	
Structure		300	
Propulsion System (Empty)		3000	
Guidance, Control, Telecommunications		700	
	MARS		**ECLIPTIC**
Propellant	4300		4560
Net Payload	800		540
Ion Stage Initial Weight	8800		8800
Support Structure on 2nd Stage (*Centaur*)	200		200
Weight in 300 nm[b] (Parking) Orbit	9000		9000

[a] Based on Atomics International and Aerojet-General estimates.
[b] Assumes a co-tangential thrust program to put spacecraft into 300-mile orbit.

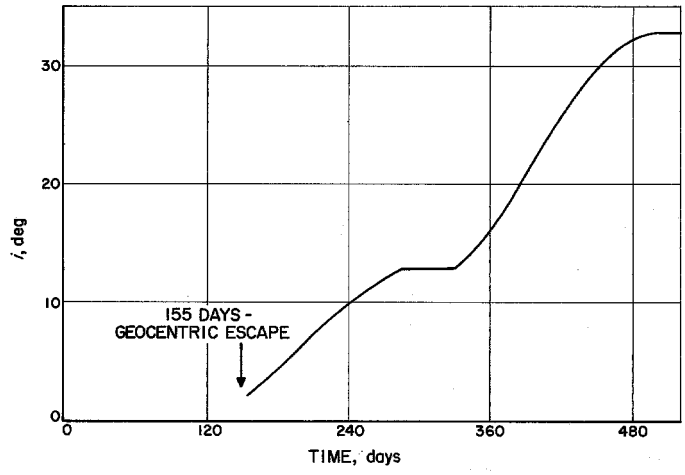


Figure 10. Inclination angle vs time

As the vehicle rises above the ecliptic plane, it will reach a point on the orbit where the velocity vector tends to pitch down toward the ecliptic (after about 295 days). Approximately 25 deg prior to this point (or 65 deg from the ascending node) the thrust unit is shut off and the vehicle merely coasts to a position approximately 65 deg from the descending node, at which time the thrust unit is restarted and the thrust vector is pointed in the opposite direction (or down from the orbit plane).

One important result of this trajectory is that, due to the thrust being added normal to the velocity vector, the energy of the orbit remains constant, which means that the period of revolution in the orbit does not change (remains at 365 days). Since the period remains unchanged, the solar longitudes of the probe and the Earth remain approximately equal. This relation is, of course, favorable to minimum communication distance for a given orbit inclination to the ecliptic plane.

Figure 10 shows the inclination of the orbit to the ecliptic plane as a function of time. The constant value of the inclination angle corresponds to the coast period. After approximately 1 year[4], the inclination is about 17 deg. There is a good possibility, however, that the angle of inclination can be increased to between 20 and 21 deg.

By reducing the power level of the reactor-converter[5] during the 40-day coasting period to about 10% of the nominal value the overall life of the propulsion system could probably be extended reliably to a 400-day operation; this period results in a 20 to 21 deg inclination angle.

Note, however, that solar latitude, rather than angle of inclination, is of importance for space science missions. Thus it would be necessary to coast an additional 90 days (see Fig. 11) before one could realize the benefit of this 20 to 21 deg angle of inclination. Note, too, that if there is no coast period, the launch date to achieve maximum solar latitude, is considerably different (estimated 4 to 6 months) for the 17 deg inclination than for the 20-to-21-deg-angle mission.

Figure 12 shows a plot of communication distance vs time. During the first year the maximum distance is approximately 20 million mi. This is shown in the isometric drawing, where the solar longitudes of the Earth and the probe are almost identical.

Figure 13 shows a plot of the angle between the vehicle axis, consequently the ion beam, and the direct line of sight; low values indicate there may be a serious communication problem. The dotted portions correspond to coast; the discontinuity arises from the reversal of the thrust direction with respect to the orbit plane.

Figure 14 gives the gross payload x 10^3 lb vs orbit inclination. The gross payload includes guidance, control, communications, and scientific payload. Figure 10 gave an inclination of 17 deg at the end of a year and 21 deg after 400 days. The corresponding gross payloads are about 1500 lb and 1200 lb respectively.

[4]Although the contracts for the reactor and power conversion system specify a life, a ground demonstration test, of 10,000 hours for acceptance, flight times have been limited arbitrarily to 1 year in the trajectory calculations as a measure in assuring a successful mission.

[5]At present the power conversion system cannot be shut down in space and later restarted.

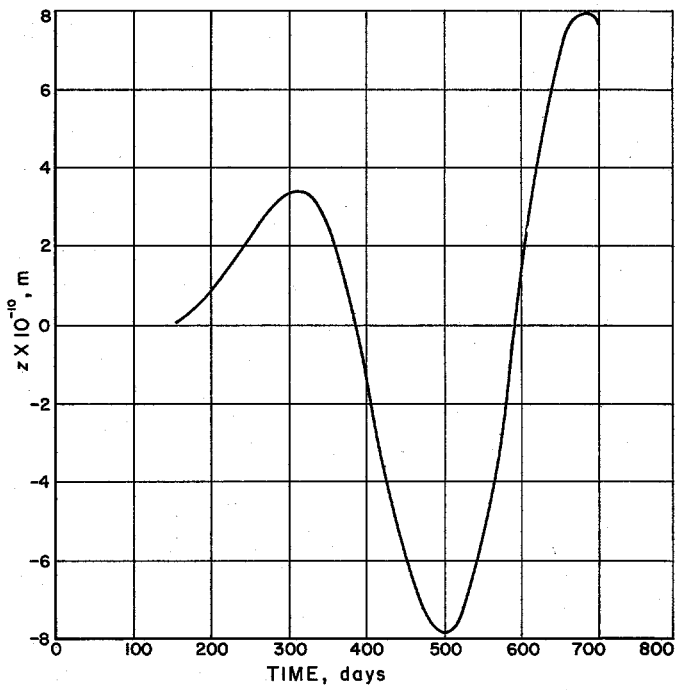


Figure 11. Distance from ecliptic vs time

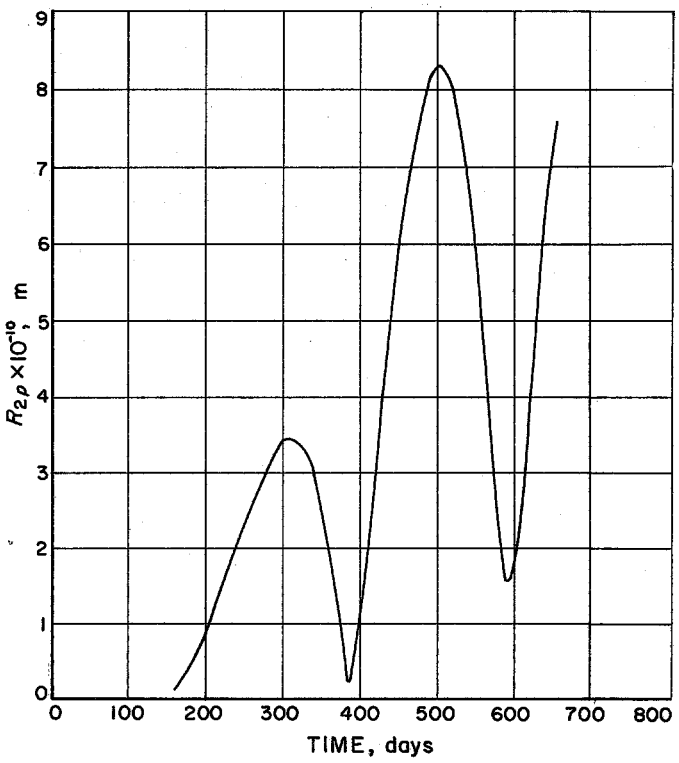


Figure 12. Communication distance vs time

Figure 15 gives gross payload vs orbit inclination to the ecliptic for various vehicles. The chemical system

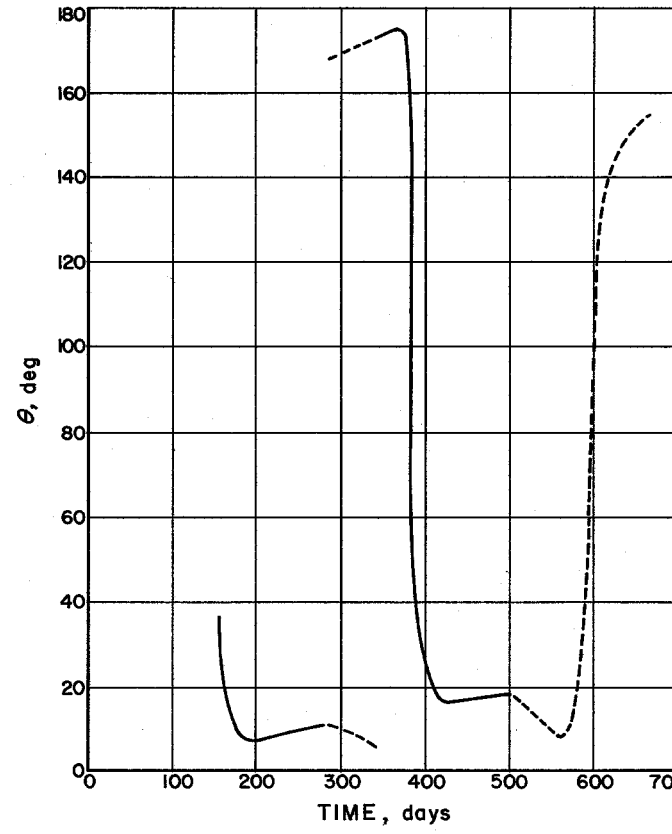


Figure 13. Angle between spacecraft axis and line of sight vs time

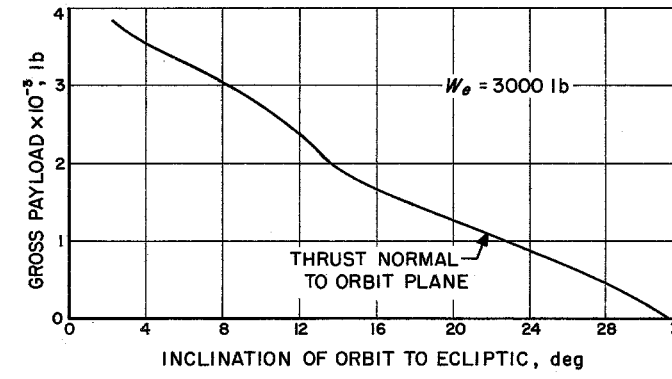


Figure 14. Gross payload vs orbit inclination to ecliptic

performances were calculated by adding their velocity impulsively to the Earth's heliocentric velocity. The ion system performances as shown in Fig. 15 were first calculated assuming the thrust was normal to the ecliptic plane continuously throughout flight. The dotted curve, on the other hand, gives the results of Dr. Melbourne's study, which utilizes the coasting period discussed previously. Prior to the coast period the results are about the same. The coast starts at an inclination of roughly

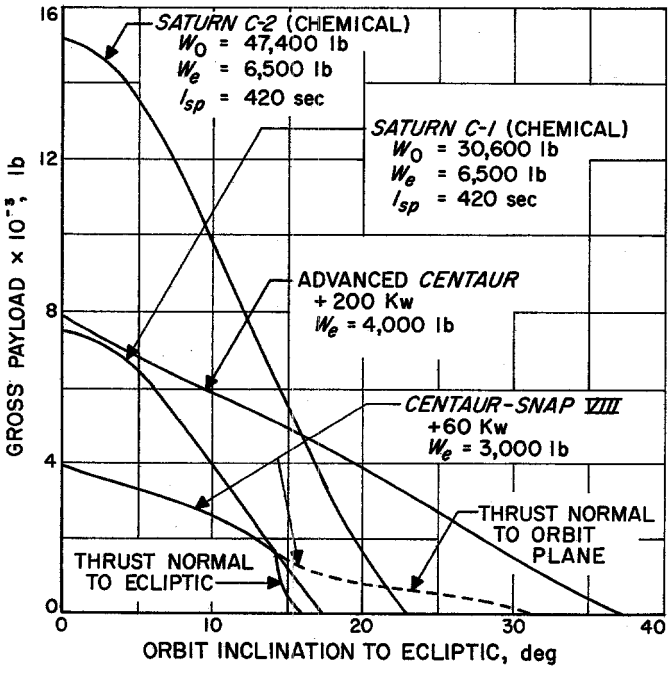


Figure 15. Gross payload vs orbit inclination to ecliptic for various vehicles

13 deg. Melbourne's results were those displayed in Figs. 4 and 5 for communication distance and look angle respectively.

It may be seen that for the very desirable high angles of inclination, i.e., greater than 15 deg, the *Centaur* with an electric spacecraft is almost competitive with the larger *Saturn C-1*. If 400 days of operation with the electric system proves feasible, the electric system would be able to reach an angle of inclination of about 21 deg with about a 1100 to 1200-lb payload, compared to about 16 deg for the *Saturn C-1*[6], with the same payload.

The performance of the *Advanced Centaur* with a 200 Kw system was computed in order to estimate the growth potential of the basic *Centaur*-electric vehicle. The specific weight of the electric propulsion system was assumed to be 20 lb/Kw; such a propulsion system could probably be available in an estimated 3 to 4 years after the *Snap-VIII* system, if funded appropriately. The *Advanced Centaur* by itself would have a payload capability of 15,000 lb in a 300-mi orbit. Calculations indicate this *Advanced Centaur* with a 200 Kw electric system would compete very favorably with even the *Saturn C-2* for very high energy missions. When sun-spot activity is very high, the inclination angles of 23 to 33 deg are desired (cf., Section II-A), such an electric-powered vehicle would provide gross payloads ranging from about 3000 to 1000 lb respectively, whereas the *Saturn C-2*[6] would be incapable of delivering a payload.

There remains much to be studied in the area of performance, both in probes of the out-of-ecliptic species and also of planetary orbiters.

[6]Performance of the *Saturn C-1* and *C-2* would, however, be improved if another small chemical propulsion stage were added to the standard vehicles; estimates of the improvement are unavailable at this time.

C. Trajectory and Performance For Mars Orbiter

W. Melbourne

For the purpose of understanding more clearly the nature of low thrust trajectories, the trajectory developed by a low-thrust vehicle in carrying out a Mars Orbiting Mission will be considered now in somewhat more detail. The mission under consideration utilizes the 60 Kw *Centaur–Snap-VIII* system and requires approximately one year to transfer from a 300-nm satellite orbit about the Earth to a 500-nm orbit about Mars.

It is convenient and appropriate to resolve the overall trajectory into segments or phases in a manner analogous to the approach taken with the space trajectories of chemical systems. Thus, Phase I is Earth-centered, Phase II is Sun centered, and Phase III is planet-centered with the usual definitions on the boundaries of each phase and the force laws included. Each phase will be considered separately.

1. Phase I

Phase I contains the geocentric spiral trajectory and its escaping asymptote. Figure 16 shows the nature of this spiral for a vehicle such as the 60 Kw *Centaur–Snap-VIII* system. This spiral was generated with a propulsion system developing a constant thrust of 0.44 lb applied in a direction tangential to the flight path. The vehicle was assumed to have an initial weight of 8800 lb which yields an initial thrust acceleration of $5 \times 10^{-5} g$. As will be explained in Section III A-4, an I_{SP} of 3000 sec was assumed which leads to a propellant expenditure of 12.7 lb/day. This vehicle commences from a circular satellite orbit of 300 nm and spirals out, completing nearly 700 turns before reaching an escape velocity at a distance of roughly 400,000 nm in 153 days. At the point of escape its velocity with respect to the Earth is about 1 Km/sec (3280 ft/sec) as compared to its initial velocity of about 8 Km/sec (about 26,000 ft/sec). The propellant consumed in escaping is 1950 lb with this thrust program.

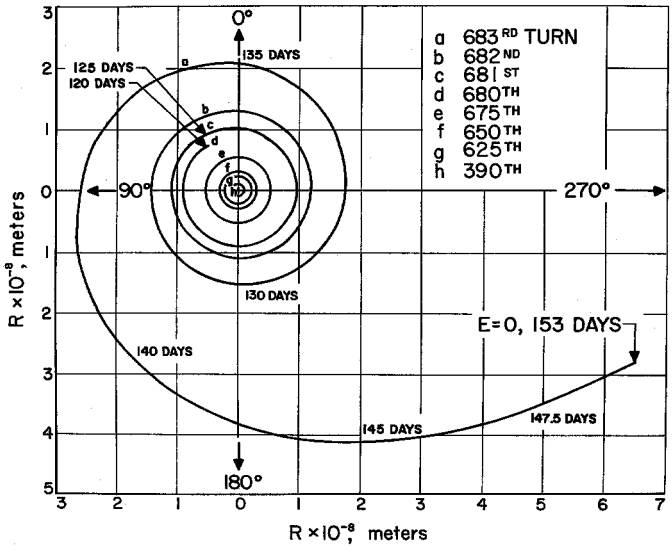


Figure 16. Earth-centered trajectory (Phase I)

For a case such as this, where (1) the thrust acceleration is very small compared with the gravitational acceleration over nearly the whole trajectory, and (2) the terminal conditions are "relatively mild," the constant tangential thrust program employed here is very nearly an optimum program in the sense that escape is effected with a minimum expenditure of propellant. That is to say, more sophisticated thrust programs net only marginal improvements in payload for this case. As will be seen shortly, this is not the case in the heliocentric transfer region where the thrust acceleration is comparable to solar gravity and the terminal conditions require a terminal maneuver with a marked departure from tangential thrust.

This spiral, then, is basic to all low thrust escape missions and is analogous to the final burning phase of chemically-powered flight trajectories. The orientation of the plane of the spiral and the direction of the asymptote will depend upon the mission. For an out-of-the-ecliptic shot, the asymptote should be essentially perpendicular to the ecliptic, whereas for the Mars mission, the asymptote, unlike chemical trajectories, is directed considerably outward from the heliocentric velocity vector of the Earth.

2. Phase II

Phase II of the trajectory is illustrated in Fig. 17. This trajectory does not precisely match the Phase I and Phase III trajectories at their interfaces; nevertheless, it gives a fairly accurate description of the motion and performance of the vehicle in this phase. The trajectory shown is idealized in that it is coplanar and assumes that Earth and Mars are massless and are travelling in circular orbits. The trajectory commences from the Earth's

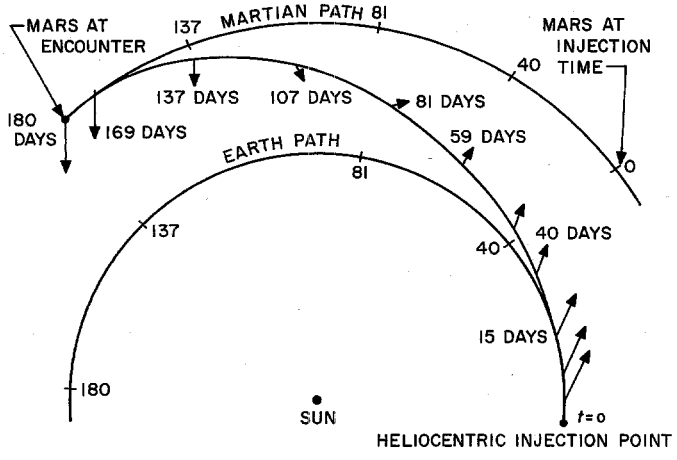


Figure 17. Heliocentric trajectory (Phase II)

orbit with the Earth's orbital speed and arrives at Mars after 180 days of heliocentric flight time. Because of the low thrust, it is necessary to encounter with Mars' orbital velocity in order to effect capture.

The thrust program employed for this trajectory is variable both in direction and magnitude. It is an optimum program in the sense that it minimizes the amount of propellant required using a propulsion system of constant power to reach a particular kinematic state in a specific time. The differential equations associated with this program were originally developed by Irving and Blum of STL. The arrows on the trajectory are indicative of the magnitude and direction of the thrust acceleration. As is readily evident, the thrust program at no point resembles a tangential program; the reason for this is primarily because the terminal point of this trajectory must possess the same kinematic conditions as Mars; these terminal conditions, then, have a pronounced effect on the thrust program over the whole heliocentric trajectory. If, instead, a trajectory were flown which simply reached Mars' orbit with maximum payload, but with no specified conditions on the terminal velocity vector, the resulting thrust program would be more clearly tangential and constant. Relative to the heliocentric radius vector the thrust acceleration vector will be seen to move through all four quadrants.

Actually, the behavior of both the magnitude and direction of the thrust acceleration vector agrees qualitatively with our intuition about the most efficient thrust program. Initially, the thrust is directed considerably outward in an attempt to place the vehicle in a region of lower gravitiation where the thrust is more effective. This factor, however, is compromised by the necessity of the vehicle to gain considerable energy which is accomplished most efficiently with the thrust in the tangential direction. This leads to the large and partially outward directed vectors shown initially. Later on, the vehicle has gained considerable energy and momentum and it enters into a quasi-coast region quite analogous to the mid-course coast trajectories of chemical rockets. During this time the thrust vector, although small, executes a considerable maneuver in which the vector is finally pointed inward. As the terminal region is reached the vehicle is again actively accelerated in an attempt to increase its energy to Mars' energy and to essentially circularize the trajectory.

Figure 18 exhibits the manner in which the energy of this vehicle varies over the flight. From 0 to 153 days, the vehicle is removing itself from the potential well of the Earth; from 153 to 333 days the vehicle transfers to Mars' energy – the small dip in the middle region is again due to a compromise between energy to be gained and the necessity to circularize the orbit. It is apparently somewhat more efficient to commence the circularization process at a slightly lower energy. Finally, the vehicle is captured by Mars and spirals into its potential well.

The mass of propellant consumed in this 180-day Phase II trajectory is 1850 lb. It will be recalled that this is about the same as that required to escape from the Earth. This may be compared to the propellant required for one non-optimum program studied which employs

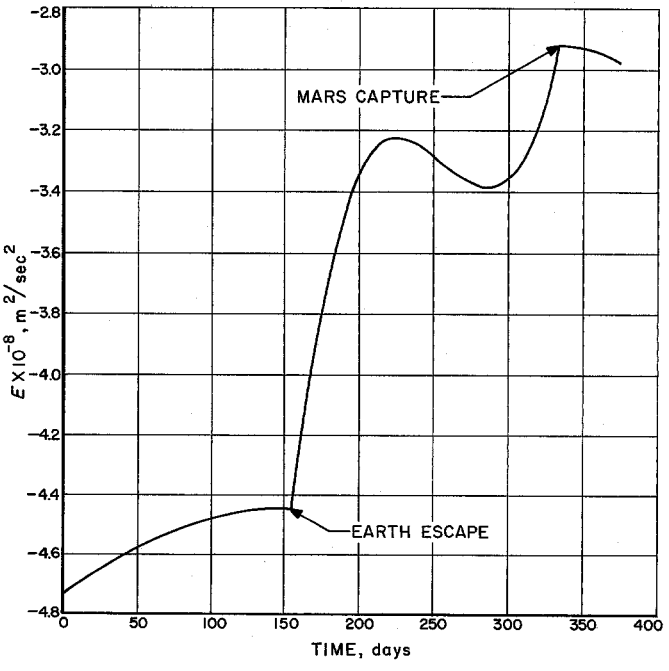


Figure 18. Spacecraft energy vs time

constant tangential thrust initially and a decelerating and circularizing terminal maneuver. The propellant required for this program for a comparable flight time is around 2500 lb.

3. *Phase III*

Figure 19 shows the Phase III trajectory and its spiral pattern about Mars. Upon being captured at a distance of about 300,000 nm and with a Mars-centered velocity of about 0.7 km/sec, the vehicle commences a tangential retro thrust again at 0.44 lb and 3000-sec I_{SP}. To spiral down to a 500-nm circular orbit about Mars requires about 100 revolutions and 40 days, which leads to a propellant consumption of about 500 lb.

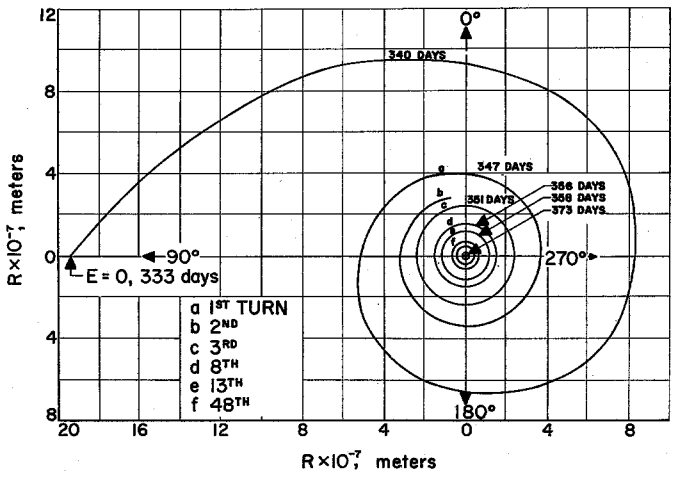


Figure 19. Mars-centered trajectory (Phase III)

For this overall mission, requiring one year for its completion, the total propellant consumed is 4300 lb. leaving a final vehicle weight of 4500 lb[7]. This figure leads to a gross payload (guidance and control, telecommunications, and scientific payload) of 1500 lb. This may be compared to the gross payloads of the *Centaur* chemical system for Mars orbits. For a 240-day flight time the *Centaur* system capability is estimated at 300 lb for a 500-nm circular orbit, and 660 lb for a 500 to 20,000 nm orbit. Thus the net payload for the electric *Centaur* is estimated at 800 lb whereas net payloads for the chemical *Centaur* are negative and zero lb for the indicated circular and elliptic orbits respectively.

No matter what trajectory modifications are tried, the propellant requirements cannot be reduced in the Mars spiral; but with encounter at perihelion 600 lb can be removed from the heliocentric propellant requirements.

[7]Performance has been based on the Irving and Blum trajectory which provides an optimum (variable I_{SP}) program. Other, more realistic, programs will be checked soon and will reduce the payload to something less than the above optimum values for the indicated flight time and power level. Note too that at this early stage in the study, payloads are relatively sensitive to basic assumptions and fluctuate appreciably.

III. MAJOR SUBSYSTEMS

A. Propulsion

The *Snap-VIII* ion propulsion system consists of a nuclear reactor heat source, a turbo-generator conversion unit, and an ion thrust unit, including the propellant feed system (See Fig. 20). The reactor is capable of producing 500 Kw of thermal power which yields a net of 60 Kw of electrical power from two 30 Kw turbo-generators. This electrical power is utilized to drive an ion engine, thus producing propulsion for the spacecraft.

Briefly, the overall system operates in the following manner. The heat generated in the reactor is transferred to NaK (a sodium potassium alloy), which is circulated through the reactor core. The NaK in turn transfers this heat to boiling mercury, which is used in a Rankine cycle conversion system.

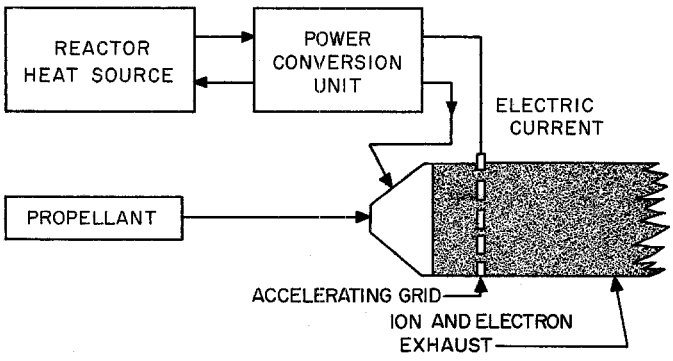


Figure 20. Schematic of ion-propulsion system

The electrical power is used to impress a high DC voltage on an accelerating grid. With cesium as a propellant, this voltage may range from about 600 to 6000 volts, depending on the specific impulse desired. The propellant atoms are ionized and accelerated by this charged grid network.

Each of these components will be discussed in more detail in the following four sections.

1. *Snap-VIII* Reactor and Shielding

D. F. Spencer

The *Snap-VIII* reactor, which is being developed by Atomics International for the AEC, is composed of a number of ½"-diameter rods fueled and moderated with 10% uranium and 90% zirconium hydride (See Fig. 21). These rods are clad[8] with stainless steel and, as mentioned above, transfer their heat to a NaK coolant. The

[8]In addition an inhibitor is used between the stainless clading and full rod to minimize the loss of hydrogen.

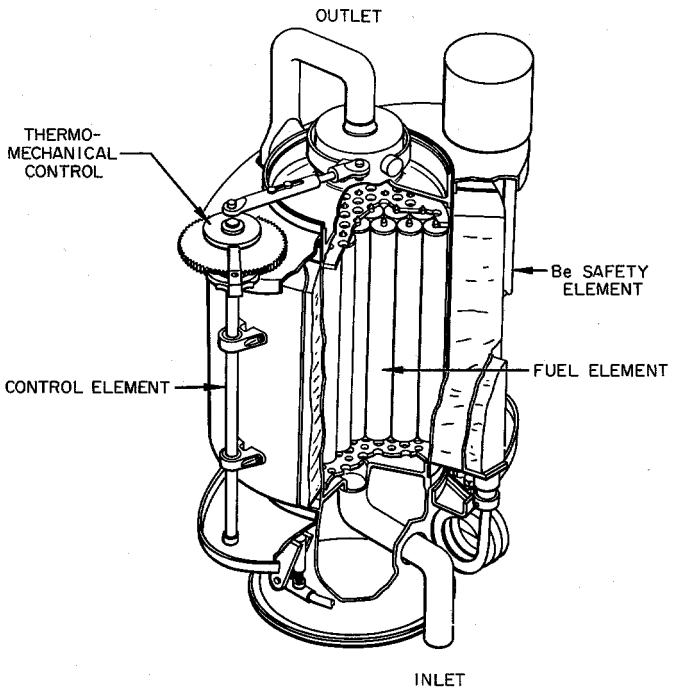


Figure 21. *Snap-VIII* reactor

maximum coolant temperature is limited to approximately 1300°F in order to prevent the disassociation of the zirconium hydride and subsequent diffusion of the hydrogen out of the moderator. The very high hydrogen concentration obtained in the hydride represents a breakthrough in reactor technology.

The reactor is controlled by four rotating beryllium drums spaced symmetrically around the core. The drums are mechanically driven into the core to increase the neutron economy and are rotated out to decrease the economy.

The utilization of such a reactor power source in a spacecraft creates a number of critical areas which are not prevalent with conventional spacecraft. Probably the most important is the political implications which might arise, due to malfunction of the system during operation, thus producing a possible radiation and public health hazard.

During the week of June 6, this topic was discussed in some detail before the Aerospace Nuclear Safety Board, an AEC Committee which is responsible for investigating the hazards associated with all nuclear space programs. They have recommended that all *Snap* systems be given approval for flight if certain criteria are met. In the case of reactor systems, the two major criteria are: (1) orbital start-up and (2) a sufficiently long-lived orbit (approximately 100 yrs), unless it can be assured that the reactor will burn up upon re-entry.

There are also a number of technical difficulties which are peculiar to the reactor-powered spacecraft. These are mainly associated with the high nuclear radiation field emanating from the reactor and the attendant shielding problems. In particular, the effects of radiation damage on sensitive elements, such as certain electronic components and the complications in performing scientific experiments to measure cosmic rays and associated radiations, are the most critical design considerations.

The former problem is due mainly to fast neutron irradiation of electronic components resulting in permanent damage to them. A particularly difficult problem is that of the back-scattering of neutrons from the radiator into unwanted regions of the spacecraft proper. However, the magnitude of this problem may be minimized by a "payload towing" technique which will be discussed later. The requirements for shielding these systems cannot be made to fit generalized conditions because it has been found that the actual function of a particular component dictates the degree of radiation resistance it must exhibit. Thus, shield design for these fast neutrons is entirely dependent on the type of circuitry used and the actual operation of a given component.

In considering the design of the instrument package, the problem of gamma ray interactions with the sensitive equipment is quite severe. The primary gammas and secondary electrons resulting from gamma interactions must be discriminated against if cosmic ray experiments which are sensible are to be performed.

This discrimination problem requires sophisticated instruments whose weights are probably not negligible with respect to other system weights. For example, if a collimator to measure angular dependence of some cosmic radiation is utilized, very large shield weights to attenuate radiations in other directions will be required for reasonable sensitivities.

One solution to the problem may be to utilize a separable instrument package which would be "towed" by the spacecraft proper. This technique, although relieving the back-scattering and direct radiation problem, introduces new problems in the spacecraft design. A brief analysis to determine an approximate separation distance between reactor and cosmic ray experiments has been made. Based on the assumption that the dose from the reactor should just equal the background dose which is to be measured, this distance is approximately 600 ft. The analysis assumes straight line motion of the gammas

through the spacecraft and attenuation by the fast neutron shield, conversion system, and propellant tank.

The basic assumption of equal doses is quite conservative (the exact degree of conservatism is unknown since the particular instruments to be used have not been selected), and thus, this distance should represent a maximum. In addition, it appears that local shielding of sensitive components, both from fast neutrons and gammas, will produce a reduction in the total shield weight required. A more thorough examination of these problems is planned as the reactor and spacecraft designs progress.

2. *Snap-VIII* Power Conversion System

J. W. Lucas

NASA has awarded Aerojet-General Corporation a contract for approximately 8 million dollars to develop the *Snap-VIII* Power Conversion System. Their Azusa facility has prime responsibility while their San Ramon division will act as consultant on reactor questions.

As shown in Fig. 22, there are two fluid loops in the conversion system. The NaK which flows in the primary loop transfers heat from the reactor to the boiler and is circulated by a centrifugal pump as shown.

A Rankine cycle has been chosen for the secondary loop. (A direct conversion system was not selected because a technological breakthrough would be required to develop such a system at the *Snap-VIII* power level

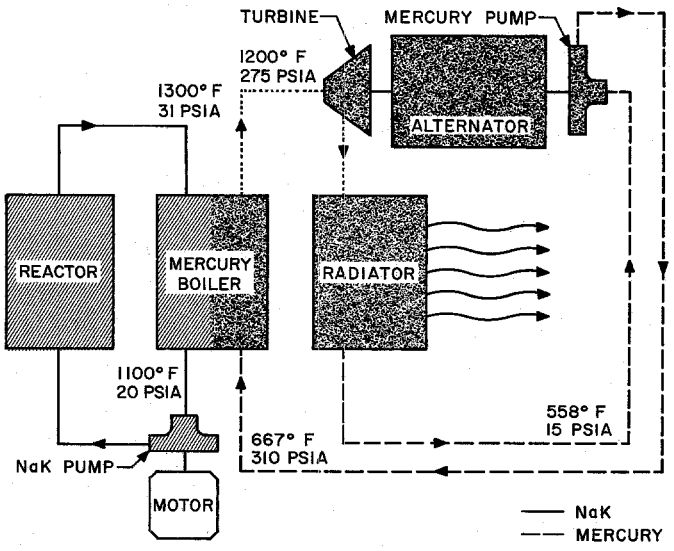


Figure 22. *Snap-VIII* power conversion system schematic diagram

and on the *Snap-VIII* time schedule.) The mercury working fluid is vaporized and superheated in a shell-and-tube counter-flow boiler. Mercury is used because its vapor pressure at the boiler exit temperature of 1200°F is not excessive.

The mercury vapor drives a multistage axial-flow turbine which was chosen to keep differential velocities between the vapor and the turbine blades low, thus minimizing blade erosion.

After expanding through the turbine, the mercury is ducted to the condenser-radiator. Here again, mercury is convenient to work with in that the density of its vapor at the exhaust pressure of 20 psia is high enough to allow the use of reasonably-sized vapor manifolds. The radiator temperature is chosen to minimize system weight and not, for example, to maximize conversion system efficiency.

After being condensed, the liquid mercury enters a centrifugal pump. Upon leaving the pump, the mercury cools the alternator, lubricates the bearings, cools the NaK pump motor and then re-enters the boiler. (This is not shown in Fig. 22.)

The turbine which rotates at 20,000 rpm drives the electric generator and mercury pump which are on the same shaft.

In Fig. 23 the radiator is shown at the upper left. Its structural aspects will be discussed in Section III-A-3. In the center are shown the mercury boiler, the mercury vapor turbine, the alternator and the very small mercury pump at the left. To give an idea of size, the boiler is 15 in. in diameter. The NaK expansion reservoir is needed to receive the NaK forced out of the primary loop when it is heated. The mercury flows through 6 parallel tubes in the boiler, drives the turbine, and is exhausted into the two manifolds which direct it into the condenser radiators. The liquid mercury returns from the radiators to the mercury pump at the lower left.

In the model in Fig. 6, the boiler and rotating unit including the control packages are represented by the cylinders and small sphere between the radiator sections.

There are a number of rather unique problems associated with the use of the *Snap-VIII* power conversion system.

1. The first of these is that of power plant start-up after the spacecraft is placed in orbit. Its equilibrium temperature just prior to start-up is expected to be below 208°F. Therefore, the low melting alloy NaK (MP = 12°F) was selected in prefer-

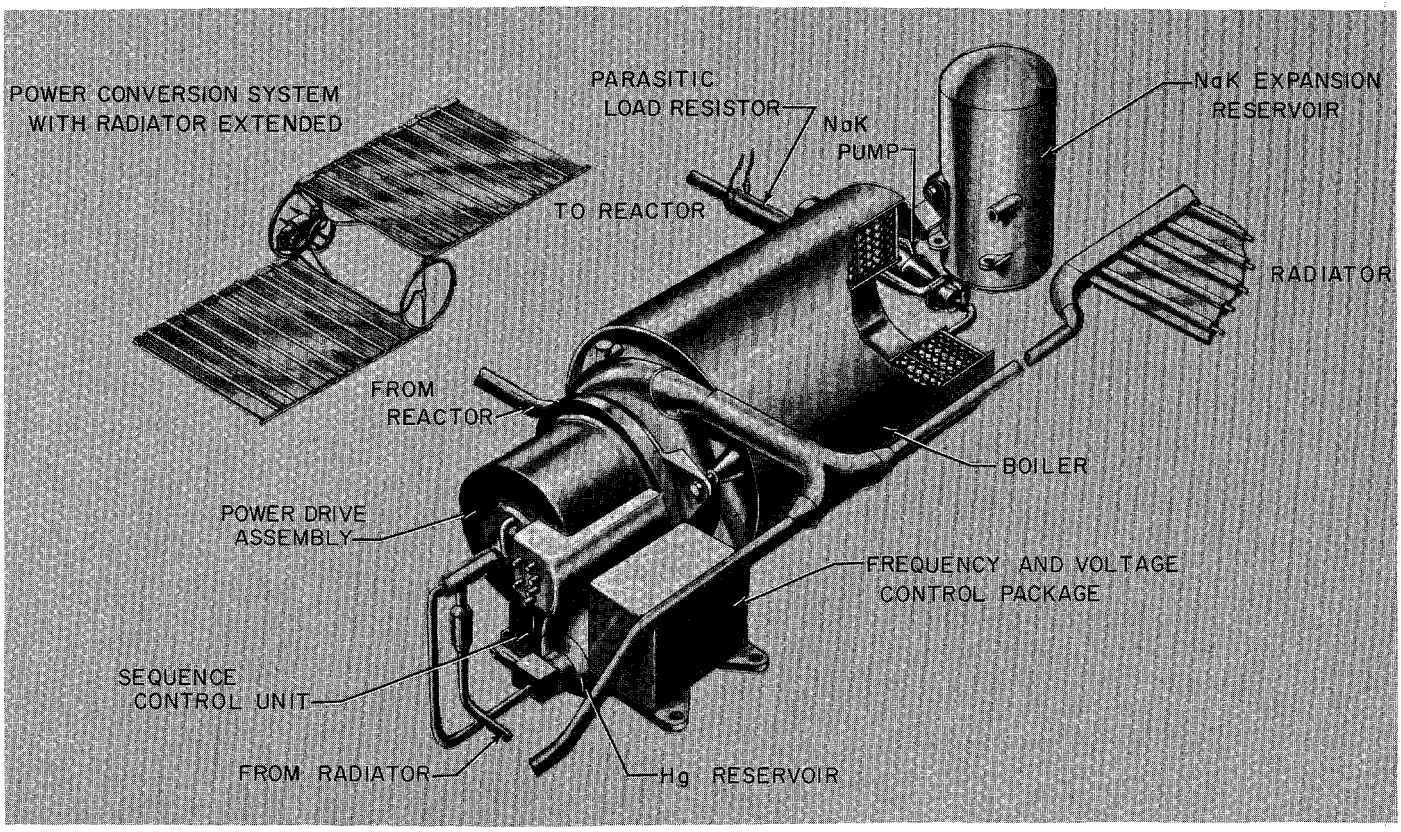


Figure 23. *Snap-VIII* power conversion system

ence to pure sodium (MP = 208°F) in order to insure that the heat transfer medium in the primary loop will be liquid at all times. Start-up in the secondary loop is more complicated because: (1) It is necessary to establish the correct direction of mercury circulation and to locate the initial high-pressure point at the mercury pump and (2) a two-phase system is involved. The mercury, stored in a separate tank until start-up, is injected under pressure at a point upstream of the boiler. A check-valve will prevent back-flow and an explosively-actuated shut-off valve separates the mercury reservoir from the rest of the system after completely discharging the fluid from its storage tank. In addition, the temperature of all system components at start-up must be high enough to prevent freezing of the working fluid at any point of the cycle. The most critical area is the radiator which has been radiating heat into space without the heat input that will soon come from the condensing mercury vapor. The radiator will, therefore, require suitable (rejectable) insulation or an auxiliary heat supply to prevent freezing and permanent attachment of the mercury as it first flows into the radiator.

2. Another unique problem is that of operation in a zero g, or more correctly, a free-fall environment. There is a twisted ribbon inside each mercury boiler tube, which swirls the liquid mercury, keeping it against the hot tube wall until it all vaporizes. Because of the resulting high centrifugal g field, free-fall is no problem here. The area of principal concern is in the condenser tubes where the vapor must be separated from the liquid. In the *Snap-VIII* system, there are three steps taken to prevent vapor lock. First, the mercury is subcooled 100°F below its boiling point in the radiator. Secondly, there is a centrifugal separation action at the end of each condenser tube. Third, there is a jet booster on the mercury pumps so that it can swallow vapor. The Tapco Division of Thompson Ramo-Wooldridge has taken motion pictures which show successful vapor/liquid separation in straight condenser tubes during a 30-sec interval of free-fall in an airplane; they believe that equilibrium conditions are established within one sec. Further tests in aircraft, as well as in rockets and satellites, are planned by Tapco.
3. There is, of course, the general problem of achieving long-life reliability of the system. One aspect

of this problem is the containment of the working fluids. To achieve this, the NaK is in one completely hermetically sealed loop and the mercury is in another. A converse problem is that of keeping any non-condensible gases out of the primary and secondary loops, such as hydrogen evaporating from the reactor fuel elements. Another problem is mercury corrosion. Mercury slowly dissolves wall material at high temperature and carries the material to lower temperature areas where it is deposited. If scale accumulates and seeps into the bearings which have a very small clearance, the system will be rendered inoperative. Previous engineering experience with mercury in large power plants has been at operating temperatures about 200°F below those of *Snap-VIII* and corrosion is expected to be much worse at the higher temperatures. On the other hand, a 2500-hr test of the *Snap-I* mercury conversion system, which operates at a temperature only 50°F below that of *Snap-VIII*, was successfully completed recently. This test was particularly encouraging because it was conducted with materials which are known to be inferior compared to newer, presently available ones. The test may not have been conservative in another respect, however, in that scale may have settled to the bottom of the lines; in a free-fall environment there should be a greater tendency for the scale to reach the bearings. High temperature mercury corrosion tests have a high priority in the Aerojet development schedule.

4. Additional unique problems include possible deleterious effects of the radiation from the reactor on the conversion system, and the reduction of unbalanced gyroscopic moments from rotating parts, including the working fluids, to levels determined by attitude control requirements.

In conclusion, the present *Snap-VIII* power conversion system development schedule calls for:

(1) Delivery of two 30 Kw systems at the end of the 1963 calendar year. These units shall have demonstrated reliable operation for 90 days in ground tests with the *Snap-VIII* reactor.

(2) Two additional 30 Kw conversion systems are to be delivered in the middle of 1965 with a proven reliability of 10,000 hr, or in excess of 400 days.

A large portion of the scheduled time is devoted to long duration reliability tests. It should be noted that the design of the system does not in any way limit its life to either 90 days or 400 days. Both these times have been selected as reasonable mileposts; subsequent testing and indicated improvements will extend the reliable life.

It is important to point out that the electrical output from the power conversion system as defined in the preliminary contract is three-phase, four-wire alternating current at 1000 cps and 43.6/75.6 v. Transformation and rectification of the electrical power will be required to supply spacecraft power for communication, guidance, instruments and controls, as well as for the electric thrust unit itself. Spacecraft weight must be allotted for the transformer/rectifier and a coolant subsystem, if the latter is required. Account must also be taken of the power loss associated with transformation/rectification. As yet, no contract has been let by NASA for this transformer/rectifier equipment.

3. Some Comments on Spacecraft Structure and Optimization Studies of the Radiator

M. E. Alper

The structural configuration of an electrically propelled spacecraft will be significantly different compared to configurations of the past. Heavy concentrated masses and very light connecting structure will present more launch dynamics problems than the Laboratory has encountered before. Structural dynamic implications will have to be considered very early in preliminary design and may govern configuration or launching sequence. In flight, forces will be low, but the rigidity and damping in the radiator or the radiator fins will also be low and dynamic problems must be anticipated.

The radiator is an intriguing part of the structure and assumes importance because of its relatively large weight and size. The number of variables which can be considered in optimizing are large, and the inter-relationships between the radiator, the rest of the structure, the conversion system, and the payload are probably more numerous than the Laboratory has faced before in the area of structures.

The radiator may or may not be part of the load-carrying structure. Temperature and pressure drop in the radiator depend on the conversion system. Configuration and weight optimization depend on the reactor shield weight, the payload, and the conversion system.

There are many variables to be considered within the radiator itself. Consider, for example, a simple tube and fin design. Tube diameter is a function of tube length and allowable pressure drop. Tube thickness depends on stress, temperature, and micrometeorite effects, either

puncture or spalling. The puncture problem is well known. Spalling caused by micrometeorite impact results in particles of the tube flaking off inside the sealed conversion system; system failure could result if particles settled in the turbine bearings and caused seizure, or clogged narrow passages and prevented flow of the working fluid.

Spalling is particularly serious, since preliminary estimates indicate that the tube thickness required to prevent spalling is from two to four times that needed to prevent puncture by a particular particle impact if the rate is the same and if no special protective measures are taken. The bumper concept of micrometeorite protection promises considerable weight saving which, as a result of the spalling criteria, may be quite important. However, information needed for good preliminary bumper design studies is not presently available, and a choice of protection schemes should be delayed until the information is available, or until a design freeze becomes mandatory.

Fin area is a function of tube spacing, fin thickness and surface emissivity; the thickness can be chosen to minimize weight or area for a given tube spacing and is limited by fabrication techniques.

Tube spacing is determined by minimizing weight, considering the total tube area exposed to micrometeorites and optimizing with header weight.

The headers can be fixed or elastically movable and are sized by flow considerations and by reliability, as far as micrometerorite-caused puncture or spalling.

Preliminary parametric studies lead to the curves in Fig. 24 which show weight, exclusive of header weight, as a function of fin thickness for two tube spacings and two values of emissivity. The study was made on a one-foot square section of radiator and the weights were calculated considering the radiation efficiency of the section and the total power that must be radiated. The minimum points will change if the tube thickness changes; this is a function of puncture or spalling design criteria and has not yet been studied. Thickness of the steel tubes used in these calculations is about 0.050-inch which gives a 90% probability for no penetrations in one year.

As a result of varying tube spacing, the envelopes of weight versus radiating area shown in Fig. 25, are obtained. Both of these curves show the marked effect of surface emissivity and present some idea of the weight that must be paid to reduce radiator area. Figure 26 shows the geometry of two different types of radiators.

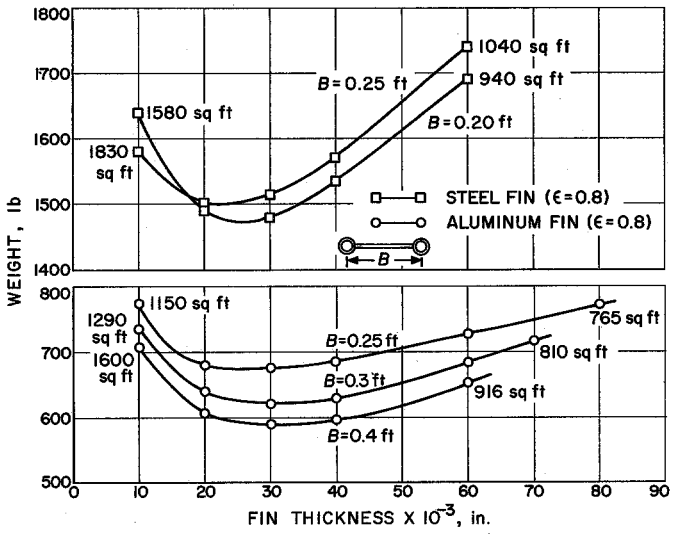


Figure 24. Weight vs fin thickness

The areas are minimum areas from Figure 25. It is worth noting that the emissivities used in this study were assumed to be available with no weight penalty, implying that the material naturally has this emissivity or that the coating required to produce it has zero weight.

It is also worth noting the relatively large tradeoff in area possible with a weight penalty that is relatively small, compared to the total spacecraft weight. This is quite important from a systems point of view, since the area saved may be far more important than the weight paid for it. Indeed, depending on the control problem caused by a very flexible radiator, spacecraft weight could conceivably decrease, as a result of the smaller area involved, even though radiator weight increased.

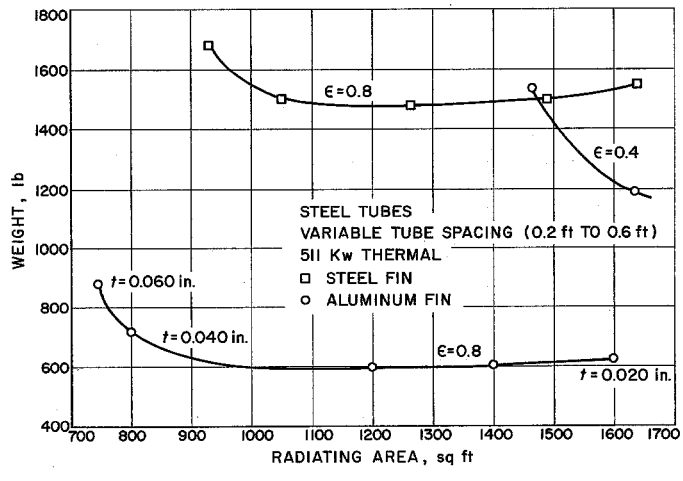
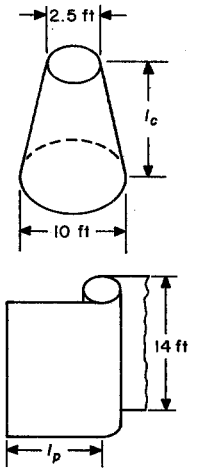


Figure 25. Radiator weight vs area



TYPE	EMISSIVITY ϵ	l_p ft	l_c ft	A sq ft
Al	0.4	31.2	89.0	1750
Al	0.8	18.8	53.5	1050
STEEL	0.8	21.4	61.0	1200
STEEL	1.0	9.8	28.4	556

Figure 26. Comparison of radiator areas for cylindrical and flat-plate configurations

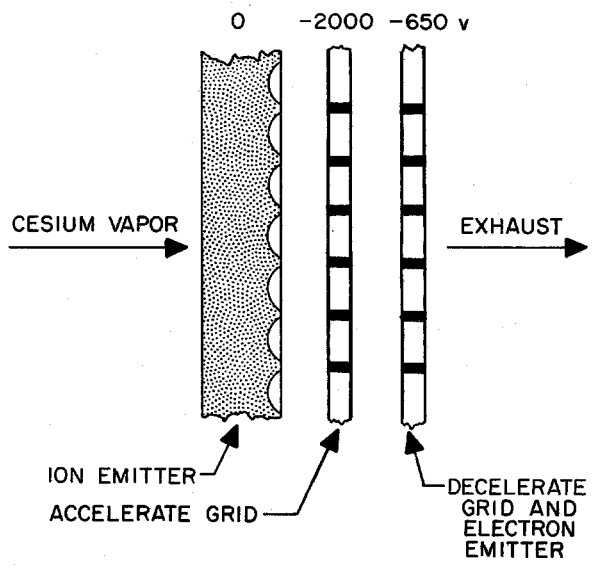


Figure 27. Ion rocket motor

The studies performed to date indicate the need for testing, possibly including testing in a total space environment, to discover the environmental effects on the surface emissivity of plain or coated materials. The desirability of developing durable, lightweight, high-emissivity coatings for aluminum or other materials is also evident.

The behavior of pressurized tubes filled with gas or liquid and subjected to micrometeorite impact must be looked at and the extent of spalling due to particle impact must be determined in some detail.

Summarizing briefly, the design of a flight radiator is going to involve more interactions and trade-offs than have been experienced with a structural subsystem in the past.

4. *Thrust Unit Capabilities and Problems*

D. G. Elliott

In an electrically-propelled spacecraft the thrust unit has the job of accelerating the propellant to a high velocity to produce thrust. Of the types of thrust units suitable or deep-space missions, the one most likely to be available first is the ion rocket motor, shown in Fig. 27. The ion motor has three main parts: a positive ion emitter, a grid assembly for accelerating the ions, and an electron emitter for neutralizing the ions. In the motor shown in Fig. 1 the ion emitter is a porous tungsten plate heated to about 2000°F. Cesium (MP = 83°F; BP = 1238°F), the most easily ionized element, enters the tungsten plate as a vapor and leaves as a current of positive ions. The accelerating, or "accel," grid draws the ions away from the emitter. At voltages which give satisfactory current densities, the resulting ion velocity is generally too high. Therefore, a decelerating, or "decel," grid is added. Finally, electrons are emitted to neutralize the space charge of the ion beam and prevent the ions from decelerating after they leave the motor.

The decel grid voltage shown in Fig. 27 corresponds to operation at a specific impulse of 3000 sec, and the accel grid voltage is a typical value for that specific impulse level. The cesium ions are first accelerated across 2000 volts, giving a current density of 5 amp/sq ft with 0.1-in. spacing between the emitter and accel grid. The ions are then decelerated to 650 volts and leave the motor at 100,000 ft/sec.

With 60 Kw input power the ion current is about 45 amps, requiring an emitter area of 9 sq ft. The corresponding 60 Kw ion motor is shown in Fig. 28. The motor is visualized as consisting of a cluster of 37 small ion motors, each 8 in. in diameter and about 4 in. long. The assembly would weigh about 200 lb.

The chief power loss[9] in an ion motor is the heat radiated from the ion emitter. This loss sets an upper limit on the thrust that can be obtained, as shown in Figure 29. This figure presents the thrust of a 60 Kw ion motor as a function of specific impulse. The maximum thrust, for the particular losses assumed, is 0.44 lb at a specific impulse of 3000 sec. At lower specific impulses the thrust

[9]Over-all engine efficiency, the ratio of ion-beam kinetic energy-to-power input, was assumed to be 50% at the indicated I_{SP} of 3000 sec.

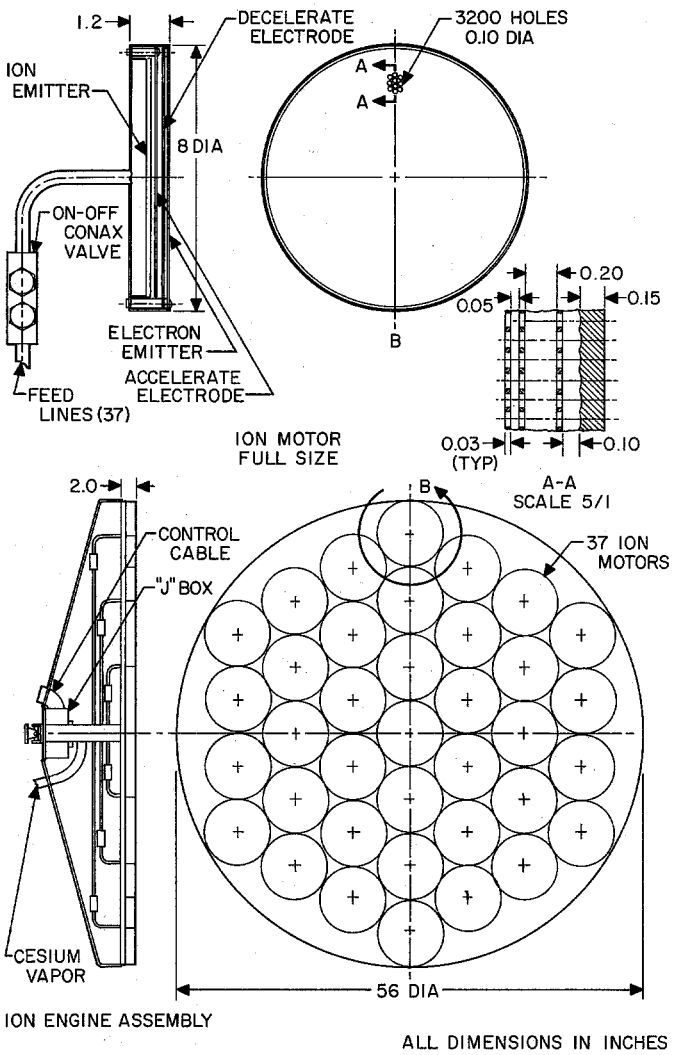


Figure 28. 60 kilowatt ion motor

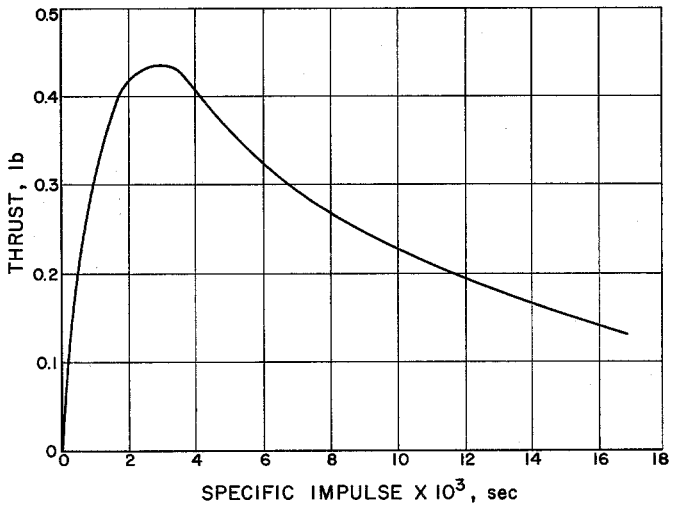


Figure 29. Predicted thrust of 60 kilowatt cesium-ion rocket motor

drops off because of heat radiated from the increasingly large emitter required, and at higher specific impulses the thrust drops off because of the fixed input power. It should be emphasized that this thrust curve is only an estimate of future capability. Those working on ion motors believe such performance is attainable, but only with considerable effort, especially in the low specific impulse range. It is not yet certain, however, that specific impulses as low as 3000 sec are really necessary for interplentary missions such as orbiting Mars. This value was found necessary for the beginning and end of an Irving and Blum variable-specific-impulse trajectory, with 60 Kw power and 365-day flight time, but other thrust programs may permit higher specific impulses.

There are three main problems to be solved before efficient, long-lived 60 Kw ion motors can be developed. The first and most fundamental is the exhaust neutralization problem, which is discussed further in Section IV-A. The second is the problem of achieving very close grid spacing for high-current density and high efficiency, a problem which is especially acute at low specific impulse. The third problem is that of achieving, and maintaining, very accurate alignment of the grids to minimize sputtering erosion and permit long life.

At the present time several laboratories have operated cesium ion motors at the 300-w level for periods of hours. Two contracts have recently been awarded for development of 1.5-Kw motors. The latter could be clustered to the 60 Kw level, but they are intended only as laboratory-demonstration units and will be too heavy and have too high a specific impulse for a *Snap-VIII* spacecraft. However, it is probably possible to develop a reasonably efficient and flyable 60 Kw ion motor by the time *Snap-VIII* is available, if work aimed directly at such a motor begins soon.

B. Guidance and Control of Low-Thrust Spacecraft

W. G. Breckenridge

The present effort in the study of guidance and control for a low-thrust spacecraft is being directed toward the development of a conceptual model of the guidance and control systems. After this model is finished, a more detailed analysis of the system requirements will be made.

1. Guidance

The guidance system is based mainly on a self-contained celestial navigation system and trajectory correction computation by perturbation theory. The system would consist of star and planet sensors, a horizon sensor, and a digital computer with stored standard trajectory data and correction equations. To reduce complexity and increase accuracy, particularly near the Earth, this system could be augmented with Earth-based orbit determination and correction computation, with a radio command link to the spacecraft.

During planet-centered escape and capture spirals, the horizon sensor can measure the angular diameter of the planet to determine altitude and establish the local vertical for an attitude reference. Celestial measurements can be used to determine the position of the spacecraft around the planet. These measurements can be compared to the stored standard values to determine the necessary corrections to the standard thrust and pitch angle programs. In an Earth escape spiral, where the direction of the departing asymptote must be closely controlled, it might be desirable, or necessary, to augment the self-contained system with ground data. The accuracy of this data will decrease as the distance from the Earth increases.

Guidance during the heliocentric portion of the trajectory would be by celestial navigation. The Sun, stars, and planets would serve as attitude references for maintaining the desired orientation of the thrust vector. The measurement of the angles between these bodies can be used to determine position. By comparing these measurements to their standard values and using perturbation theory, the thrust magnitude and orientation corrections can be determined.

Inertial instruments such as accelerometers and gyros will be of only limited use. Use of accelerometers is doubtful because of the extremely low acceleration levels, and would require considerable advances in the state-of-the-art. Gyros could be useful, particularly in the escape and capture spirals, but might be confined to those periods because of limited lifetime and the long flight time for low-thrust trajectories.

2. Attitude Control

The attitude control system for the spacecraft would rely on the sensors for position reference, and stabilization would be achieved with gyros or passive compensation. Work is proceeding on a Sun-star attitude control system, and work has been done on horizon sensor systems.

The method of actuation could be one of several: reaction wheels, gas jets, or gimballed engines. The reaction wheels would be useful in moving from one orientation to another and for stabilization, but could not cope with external disturbances such as solar pressure and micrometeorite hits, since the imparted angular momentum would accumulate in the wheels and would require removal by gas jets or engine gimballing.

Gas jets, over the long flight time, may require an excessive amount of propellant because of the limit cycle operation, but have the advantage of releasing the angular momentum accumulated by the spacecraft.

Gimballing the engines has possibilities because it permits the release of the accumulated momentum, and it requires no additional energy source.

The choice of one of the three, or a hybrid system, would depend on a more complete design of the attitude control system.

One difficulty encountered in the attitude control system is that of the angular momentum of the rotating power conversion equipment. This angular momentum produces gyroscopic reactions to the attitude control torques. It is recommended that the design of the power conversion system minimize this angular momentum. A more complete attitude control study will show whether any residual angular momentum can be tolerated or must be trimmed out with vernier fly-wheels.

C. Telecommunicating From An Ion-Propelled Vehicle

W. C. Pilkington, C. T. Stelzried, and W. H. Wells

This Report discusses briefly the problems in communicating from an ion-propelled vehicle. The possible difficulties are due to the high temperatures and radiation fluxes from the nuclear reactor power supply and the plasma resonance phenomena of the ionized exhaust stream. Several novel communicating schemes are mentioned. Preliminary recommendations are made for the first vehicle.

The characteristics assumed for the vehicle are rather extreme in many ways. The temperature of the heat radiator on the vehicle is estimated to be 400°C. The vehicle, of course, will be somewhat cooler in places as radiation from part of the vehicle can keep the temperature down to perhaps 100°C, or even lower, and refrigerated cooling can be used if a considerable amount of heat must be removed. However, to pump heat by refrigeration from a temperature such as 100°C to the heat radiator temperature of 400°C uses excessive electrical power. Most present day components can operate at 100°C and many vacuum tubes and some other components, excluding most but not all semiconductors, can operate at temperatures as high as 250°C. Various laboratories have, in the research and development phase, components which operate as high as 400°C.

The nuclear reactor on board the vehicle radiates very high neutron and gamma fluxes. Estimates vary of what these fluxes will be at the critical electronic components, but the fluxes at the exterior of the nuclear reactor itself may be the order of 10^{18} fast neutrons/sq cm/year and 10^{12} Roentgen/year of 4 Mev gamma. It is expected that shielding will reduce these levels to about the order of 10^{12} fast neutrons/sq cm/year and 10^{7} Roentgen/year respectively. Even these levels preclude the use of certain instruments and semiconductor devices near the reactor, and may require these relatively sensitive devices to be removed a considerable distance from the vehicle. Vacuum tubes, however, would be quite capable of withstanding the shielded values previously listed. Vacuum tubes sometimes have glass-to-metal seal problems at 5×10^{14} fast neutrons/sq cm and at very high gamma levels. Special resistors, capacitors, and transformers would also survive at levels up to 10^{17} fast neutrons/sq cm.

Another significant difference between an ion-propelled vehicle and a conventional one is that the propelling ion beam is on throughout almost the entire flight. For the thrust and emitter area initially contemplated, an ion density of about 10^{10} ions/cc will result. The plasma frequency will then be about 1 Kmc. Any RF radiation below this frequency incident on the beam would be absorbed and reflected by the beam; thus if the transmission frequency is below the plasma frequency, a prerequisite for adequate communication is the non-intersection of ion and antenna patterns. Preliminary calculations indicate that ion beam spreading from electrical forces will be small, and that the geometry of ion ejection can be so controlled that the ion beam angle from this effect would also be reasonably small, i.e., a few degrees. RF noise generation by the beam at frequencies of interest, is possible, but it is expected that any noise generation would occur at frequencies below 2 times the plasma frequency. The magnitude of this noise would not exceed a small fraction of a watt, but a fraction of a watt could be disastrous at the receiver frequency. Above the plasma frequency an ion beam would look like a dielectric, and thus would distort or bend any radio waves passing through it. It is felt that both the transmission frequency from the vehicle and the receiver frequency of the vehicle should be well above plasma frequency for the ion beam. The plasma frequency at the exhaust is about 1 Kmc. The density of the beam goes down as a function of distance from the vehicle, due to spreading effects, so that the plasma frequency which varies with density to the ½ power also goes down. This results in an overlap or smear of the resonant peaks to give noise over an extremely broad frequency range. Due to cylin- plasma frequency $\div \sqrt{2}$. Additional harmonics (Dattner peaks), whose amplitudes decrease with the order of the harmonic, result in a distribution of resonances clustered about the plasma frequency as shown in Fig. 30. Aside from noise considerations, it is felt that the antenna must never point through the beam, due to refraction of the antenna pattern.

Another significant new feature of nuclear-electric vehicles is quite adequate power for communication. Sixty Kw is the power available for all uses, a quite significant power. Vehicles in the past have had nowhere

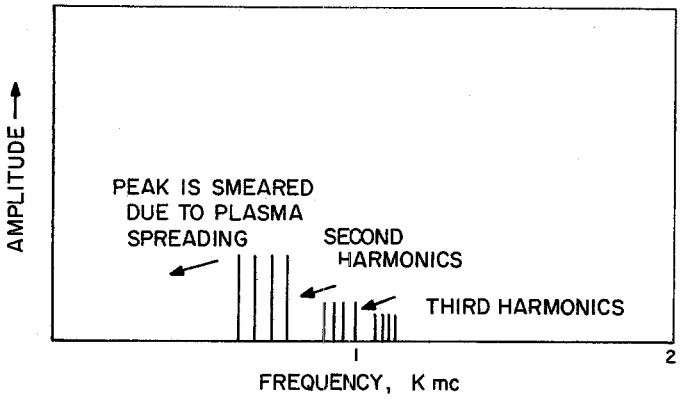


Figure 30. Smeared plasma resonances

near this power available, and have had to rely on low-energy devices to a much greater extent than would be necessary in the case where a nuclear reactor is carried aboard.

The following description will identify the type of transmitters and receivers that might be carried for a first mission where reliability and minimum complications are desired, and the range would be approximately 2×10^8 Km or a little over 10^8 mi. The antenna would be approximately 3 m or 10 ft in diameter, a steerable dish weighing roughly 50 Kg or 100 lb. A restriction would be put on the antenna and the vehicle such that this antenna would never point through the ion beam. The antenna surface and directivity must be adequate to use it effectively at the frequencies of the transmitter and receiver. The transmitter would radiate approximately 100 watts at about 2 Kmc. It would be an all-tube unit to avoid the problems of irradiation and heating, since there is no significant power limitation problem or even efficiency reduction when the power input reaches the order of a few hundred watts. The transmitter would be mounted in reasonable proximity to the antenna. There would be a receiver at about 2 Kmc, also mounted close to the antenna. It would probably not be an ultra-sensitive receiver as it is not in an extremely quiet location, and because adequate power is assumed to exist at the transmitter on the earth. The noise contributions to this receiver would be significant compared to those in a vehicle in a cooler environment, but not overwhelming. It is estimated that the thermal temperature of the ion beam would be on the order of 3000°K, depending on the temperature and electric potential of the electron emitter, but the antenna would never look at this beam, and only sidelobe effects would thus be seen. The receiver frequency would presumably be well above the plasma frequency, and thus plasma resonances would probably not contribute significant noise to the receiver.

It is of interest to discuss briefly some novel features which might be employed in later vehicles. Figure 31 shows a configuration of an ion motor and antenna which would permit completely free choice of the thrust and antenna directions without interference. The ion-emitting surface would be divided between two motors as shown in the figure. A divergence of a few degrees between the axes of the two motors compensates for beam spreading and allows the antenna to look between them. The motors are pivoted on the axis AA through the CM. The antenna presumably points at the Earth. Then a rotation about AA and a roll about the vehicle axis allows the thrust vector to point anywhere.

Many unusual communication methods or devices have been given a preliminary evaluation in this report period. A few are commented on here. It has been suggested that the ion propulsion beam itself be used as a light source for tracking. Calculations, which assume only a single photon at the tracking wave length for each ion emitted, indicates about 25 w of light would be available. This would only amount to approximately 1 photon per sq m/h at Earth extreme range and thus would be useless for tracking. If the beam were sunlight-activated, as is probable, there would be considerably more transitions, but it is very unlikely that enough transition would be available to track the vehicle to its ultimate range, using the light as a signal.

It has also been suggested that energy might be coupled out of the traveling ion beam with a cavity. For present ion engines, the size of the beam is much too large to get a usable frequency from a cavity surrounding the beam; thus this method is not usable at the present time.

The major limitation of the transmitter power using conventional techniques is the dissipation of the heat which is generated in the tubes. Approximately two-

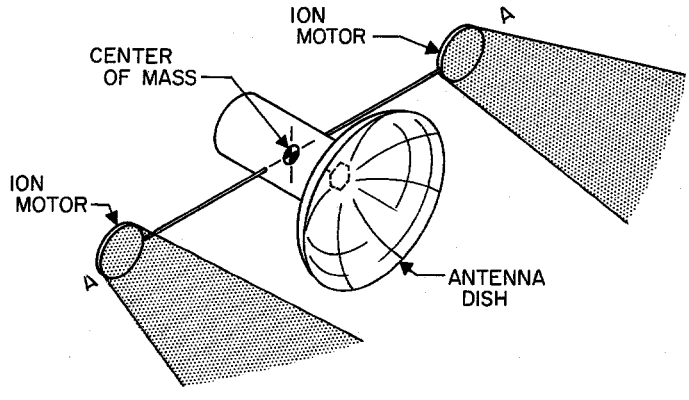


Figure 31. Possible ion-motor-antenna configuration

thirds of their total power input is dissipated as heat; only one-third is radiated as RF energy. To pump this heat to 400°C or to make a tube that will operate at 400°C are both feasible but expensive. If the beam of a high-powered tube were allowed to escape from the vehicle, rather than be collected (as in a klystron), this would reduce the cooling problem. The electrons could escape from the vehicle permanently, (the ion motor can compensate for the charge) or impinge on the radiator or other portions of the vehicle. If they hit the vehicle, the temperature at which they give up their energy would be that of the surface they hit; thus it should be quite possible to pump the heat of such a tube directly to the radiator surface without a refrigerator. A little reflection regarding the Faraday ice-pail effect indicates that an electron beam which escapes the vehicle, either permanently or temporarily through a small hole, is electrically equivalent to catching the beam on a plate at the potential of vehicle ground, as far as *internal* circuitry is concerned.

It might also be possible to use all the electrical power generated by the nuclear reactor for communication by switching at intervals from ion-beam propulsion to data transmission, using a very powerful electrical transmitter. This would not be done in an early vehicle, but might have application as the communications requirements of vehicles using ion propulsion become more severe.

Another new communication method might be the use of a light maser, both for transmitting and receiving. These devices are, of course, quite new and cannot yet be evaluated in detail for this use.

Another of the newer ideas that might be used in future vehicles is a plasma antenna. Calculations have shown that a refractive lens antenna producel by emission of a plasma from a vehicle could give an antenna of 50 meters diameter for 60 hours with the use of approximately 1 Kg of cesium, if the transmitter frequency were approximately 100 mc. This bears further investigation and has interesting possibilities for fairly low-frequency communication where continuous communication is not desired. Too much cesium is required for high frequency. Perhaps in the future it might be possible to confine a plasma such that material need not be continually ejected. At the present time, for frequencies below 400 mc, large lens antennas could conceivably be formed by the slow, continuous emission of ions in the proper density pattern.

The desire for a very reliable communication system probably limits the final choice for early vehicles to that mentioned earlier. If approximately 100 w of RF were radiated from a 3-m (10-ft) lightweight parabolic antenna aboard the vehicle, and received on earth by a 26-m (85-ft) antenna at a noise temperature of 50°K, a band width of approximately 4×10^4 cps could be achieved for a 10 db signal to noise ratio. If the vehicle antenna were 4 ft, the band width would be 6×10^3 cps. This is a very reasonable communication band width for early experiments and could be increased in the future by increasing the diameter of the transmitting dish, by increasing the power from 100 w, or by using some more unusual communication method, such as one of those mentioned earlier in the report. Communication from the Earth to the vehicle should be no problem as at least 10,000 w would be available on Earth with the same size antennas in both directions. Thus, to achieve the same band width in the reverse direction, the receiver temperature could be as high as 5000°K, and it is believed that a band width of 10^4 cps would be adequate in the link from the Earth to the vehicle.

In conclusion, it appears that communication from an ion-propelled vehicle is quite practical and can be accomplished with conventional techniques. There will be certain restrictions and problems, such as temperature, nuclear fluxes, and avoiding the ion beam. There will be additional problems if very high information densities are desired from the vehicle, but the ion-propelled vehicle with a nuclear reactor affords the possibility of using increased transmitter power due to the availability of relatively large amounts of electrical power aboard the vehicle.

IV. FEASIBILITY, THE DEVELOPMENT PLAN, AND SUMMARY

A. Status of Charge Neutralization Problem

R. V. Meghreblian

1. Introduction

The problem of the charge neutralization of an ion beam to be used in an electric propulsion system has received the attention of a number of investigators. Of particular significance are the experimental investigations conducted at the Lewis Research Center, the analyses conducted by STL and the combined analysis and experiments of the Hughes Aircraft Co. None of these investigations have been sufficiently general, however, to yield conclusive results or a clear-cut understanding of the problem. On the one hand, the analyses have been limited to one-dimensional systems, so that the results are not entirely realistic. On the other hand, the experiments which have been conducted in vacuum tanks, although three-dimensional, have been handicapped by an incomplete knowledge of the boundary conditions at the tank walls.

The present report outlines a theoretical study of the overall problem which has been initiated at the Laboratory in an attempt to establish the essential features of the problem and hopefully to obtain some results which will be helpful in interpreting experiments and predicting thrust unit performance. The clarification of the problem is considered a principal objective of this analysis. Although a number of specific features of the ion source, the accelerator, and the ion beam, have been studied in the past, it is believed that these investigations have been too restrictive and limited in purpose. It is not clear, even at this stage, just what are the major questions to be considered, much less how to resolve them. In the past, attention has been directed toward detailed aspects of accelerator focussing, beam spreading and plasma oscillations, and the effect of space charge limitations in the ion beam on the net thrust. These problems may indeed be crucial to the operation of the ion engine, but not until a generalized formulation of the problem has been accomplished will it be possible to delineate the important effects.

2. Analysis

The problem is formulated in terms of the three-dimensional charge distribution about a body in free space which ejects both positive and negative charges (ions and electrons) in some prescribed manner. These charged particles give rise to space charges which will influence greatly the paths of all ejected particles and will also polarize the surface of the body so that the forces acting on the body differ greatly from the case in which only neutral particles are ejected. The goal of the calculation then is to obtain an expression for the

sum total of all forces acting on the body under due consideration of space charge effects. To this end, the steady state distribution functions $f_j\ (\vec{r},\vec{v})$ are determined which give the density of particles of type j in phase space. The balance relations for these functions are stated with the aid of the Vlasov, or collisionless Boltzmann, equation:

$$(\vec{v} \cdot \nabla_{\vec{r}} + \frac{1}{m_j} \vec{F}_j \cdot \nabla_{\vec{v}})\ f_j\ (\vec{r},\vec{v}) = 0$$

where j denotes ions or electrons and m_j is the mass of particle of type j. The forces F_j on the particles are given by

$$\vec{F}_j = \pm e \nabla \phi - \frac{m_j}{m_s} \vec{T}$$

where m_s is the mass of the body, and the potential ϕ is the solution to Poisson's equation

$$\nabla^2 \phi\ (\vec{r}) = -4\pi e \int d\vec{v}\ [f_i\ (\vec{r},\vec{v}) - f_e\ (\vec{r},\vec{v})]$$

The net thrust $\vec{T}$ on the body is given by

$$\vec{T} = \frac{1}{2} \int \triangle \phi\ (\triangle \phi \cdot d\vec{S}) - \int \vec{v} \cdot d\vec{S} \int \vec{v}\, d\vec{v}\ (m_i f_i + m_e f_e)$$

The first term on the right-hand side of this equation is the contribution to the thrust due to the polarization of the outer surface of the body by the space charge. The second term is the usual momentum transfer to the body by all particles leaving and entering.

The above system of equations is solved with the boundary conditions that $\phi\ (\infty) = 0$, ϕ (body surface) $= a$, and that the net charge on the body is zero, that $f_j\ (\infty, \vec{v}) = 0$, and that the velocity and spatially dependent currents emitted by the body are some pair of known functions.

Since this set of equations yield a pair of non-linear integro differential equations, there is little hope for obtaining an analytical solution, and one must resort to numerical methods. The non-linear coupling between the particle densities and the potential is opened by forcing a solution through successive approximations. Given an approximation to the potential, particle densities are calculated from the Vlasov equation and integrated over velocity space to determine net space charge and a new approximation to the potential from Poisson's equation. Acceleration to the origin of coordinates (the $\vec{T}$ term in the force equation) is neglected. After convergence of the iterative process, the thrust is evaluated by integration over velocity space and the surface of the body. The successive steps are handled semi-automatically in the IBM type 704 EDPM.

Particle densities are obtained by a numerical method based upon the characteristics of the Vlasov equation which are, of course, the trajectories of the particles. Poisson's equation is solved numerically by a relaxation process.

The particular system under study is a sphere in free space which emits electrons and positive ions at steady state. The ejected particles from the sphere are assumed to leave in an axially symmetric fashion so that the analytical problem may be represented by a four-dimensional phase space (having coordinates r, θ,v_θ and v_t). It is expected that the determination of the forces on this idealized spacecraft will reveal the essential features of the real problem and thereby suggest the proper choice of boundary conditions to be used in vacuum tank experiments, and how one is to interpret the results of such experiments in terms of free space conditions.

At present the problem is being programmed on the IBM 204. It is hoped that within 2 to 3 months results will be forthcoming and some recommendations can be made.

3. Experiment

The results of this study should be helpful also in the design of a feasibility test in free space, if one appears to be required. In anticipation of this requirement, two types of satellite experiments have been

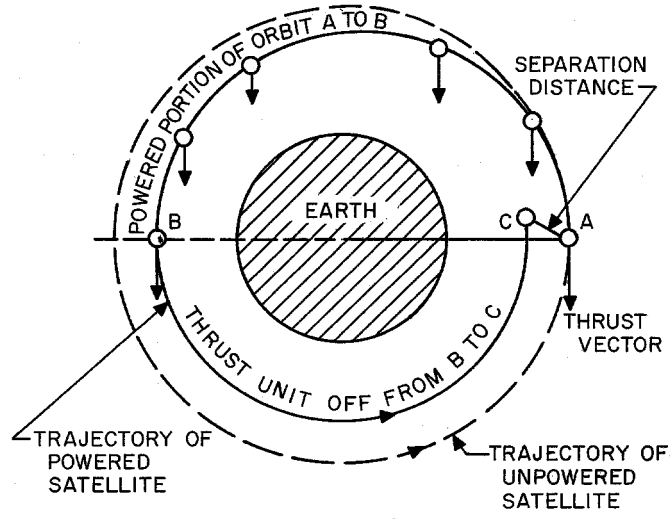


Figure 32. Thrust program and orbit for separated satellite experiment

considered. One involves the use of a small battery-powered thrust unit to despin a satellite, the analysis of the spin rate yielding information about the net forces on the body. The other proposal involves the study of the separation between two satellites starting from the same orbit, only one of which is powered by a thrust unit. Detailed trajectory analyses indicate that the maximum separation per revolution is obtained if the powered unit is fired during only one-half the orbit (see Fig. 32). If the powered satellite is initially stabilized at some known altitude in a fixed inertial frame, then for an acceleration of $10^{-5}g$ the two bodies will separate about two miles in 10 revolutions (see Fig. 33). This separation s_n can be detected by both optical and radar techniques. The separation distance measured by these means, when combined with internal thrust mount measurements, should yield a reliable determination of the net forces on the body.

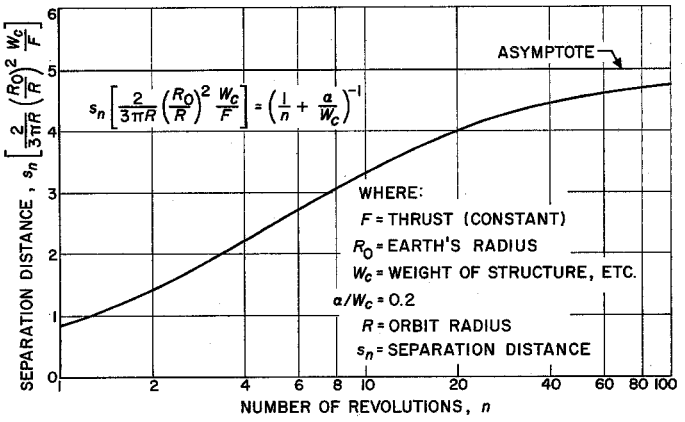


Figure 33. Separation distance s_n as function of number of revolutions for battery weight proportional to total time in orbit

B. Development Plan For Advanced Propulsion

J. Paulson

There are four pertinent phases in the development and utilization of any subsystem for space exploration. The first phase is supporting research, the second is advanced development, the third is system integration and development, and the last is mission utilization.

The supporting research phase on thrust units should go on continuously and, in this field, should concentrate on the problems associated with electrothermal propulsion (arcjet), electrostatic propulsion (ion and colloid), electromagnetic (MHD) and direct nuclear propulsion. As it is impossible to schedule breakthroughs, we can only anticipate the time at which sufficient research is accomplished to start advanced development and the engineering necessary to permit these ideas to be used in a spacecraft. At the present time, advanced development on cesium-ion propulsion systems has already been initiated. Some of the electrothermal devices are also coming out of the research stage, and demonstration engines of both types are being planned or designed as precursors of the flight units.

The present intuitive feeling is that MHD devices require more research and advanced development should probably not be started on these devices until late 1962 or 1963. The nuclear propulsion devices (Rover) are making significant progress and might well be available around 1965 or 1966. However, the programs are slanted toward a very high thrust booster stage or a relatively high thrust second stage and appear less applicable as spacecraft propulsion.

The reactor heat sources require long (5 years typical) lead times and a fairly heavy supporting research program which strives to produce lighter weight systems through higher power densities in the core, and operation at higher temperatures and maximum thermal cycle efficiency. One important problem area which should be stressed is the investigation of operating such devices at reduced power levels; operation below their nominal rating would extend overall operating life and yet provide sufficient power for scientific experiments and telecommunications after arriving in final orbit or at the spacecraft's destination. Another area worthy of research effort, that of shut-down and re-start capabilities, would provide significant advantages for certain missions. For example, if the propulsion system could be shut down during much of the heliocentric (midcourse) trajectory of a Mars orbiter mission, the overall life, or reliability, of the power plant could be increased significantly without decreasing the payload excessively. So too, the mission out of the ecliptic could be improved for power-limited systems with propulsion restart (See section II-B).

The power conversion systems supporting research should continue along the lines of higher efficiency and lighter weight systems and should concentrate on the material and alkali-metal-coolant fields. The higher temperature operating problems and the radiator problems are fields which require much investigation. It is already evident that more advanced conversion systems will operate at much higher operating temperatures and thus pose extremely severe material and design problems for research.

In all the above areas, reliability and lifetime are of importance and will undoubtedly necessitate the expenditure of large sums of money by the nation. The improvement in these areas should be studied and pushed to the limit.

The second phase is the advanced development program. In this program not only are laboratory tests made on subsystems but several flight experiments should also be included. The complete system requirements should be introduced at this time to shape the design and development of these subsystems.

At the present time the advanced development period for the cesium ion propulsion system has started and should continue through mid-1963. Space tests with a small ion motor to measure ion beam neutralization should be undertaken as quickly as possible because a question of technical feasibility is involved, i.e., buildup of space charge around the spacecraft may prevent the development of thrust by the ion engine. In this or subsequent flight tests, the effect of the beam on components in the spacecraft and on the ability to communicate with an ion-powered vehicle, should also be examined. This program should culminate with the basic design of a subsystem that can be used to optimize a spacecraft for specific missions; other propulsion devices than the ion motor should go through a similar series of tests before they are utilized.

Advanced development work on reactor heat sources and power conversion systems has already been initiated. Just as *Snap-X* is a scaled-down version of the *Snap-II* reactor, *Snap-VIII* is a scaled-up version; thus, development results from the older *Snap-II* and *Snap-X* programs should be invaluable in solving many of our *Snap-VIII* development problems. In reality, if the flight test program for the two early reactors is planned properly, component flight-testing of the follow-on *Snap-VIII* reactor should be obviated, and its first flight test should consist of a propulsion system integrated flight experiment.

It is highly probable, as work progresses, that ground experiments, and possibly flight experiments, must be performed to determine the effect on the radiator of its unusual operating conditions and the space environment. For example, little is known about long-term, high-temperature creep for relatively light, flimsy structures containing dissimilar materials, nor the effect on radiator emissivity of high-vacuum, ultraviolet and particle radiation, and of high-velocity meteoroids. It is evident, too, that penetration or spalling of the radiator by meteoroids should be investigated as thoroughly as possible in the immediate future because of the dominant influence on the design and strong effect on vehicle payload capability.

It is currently planned to fly the first conversion system in a spacecraft integration experiment. Contractually, the system is to be available, with a 90-day reliability, late in 1963; by 1965 it should be available with a 10,000-hr (414 day) demonstrated reliability.

The third phase of effort is the system integration development. NASA has made no commitment for this phase of work as of this date. If the *Snap-VIII* propulsion development schedule were the controlling factor, system integration experiments could start in early 1964; however, several other factors may introduce delays into this schedule.

System integration tests should probably start with highly instrumented flight tests of technological payloads in a spacecraft powered by a *Snap-VIII*-ion system with 90-day demonstrated reliability. If the complete propulsion system functioned satisfactorily, the flight would be continued beyond the 90 days in an effort to achieve hyperbolic escape velocity and simulate as much as possible the flight of the ultimate scientific mission itself. It is probably feasible to limit system integration experiments to one or two flight tests, although it would be very desirable to have more for a system as unconventional as this.

If manpower and funding were not a limiting factor, technical considerations alone influenced the schedule, and the propulsion development program were successful, the first full-scale mission with an ion system could be accomplished late in 1966. The first mission will probably be an out-of-the-ecliptic probe, a deep space probe within the ecliptic, or a Mars orbiter mission. Because the first propulsion system will not have restart capability, the extra-ecliptic probe would be confined to an angle of inclination of approximately 15 to 17 deg.

As new energy sources, power conversion systems and propulsion devices became available, similar development plans should be made. At this time it is reasonable to expect that after 1965, 300 Kw and 1 Mw power systems should be available for development. Note that the Air Force recently let a study contract to AiResearch and Aerojet-Nucleonics for a 300 Kw reactor-turbo-generator system; the Air Force hopes to have a tested prototype available by 1968. Interest and activity in the MHD plasma propulsion systems have been stimulated in recent months and one can expect to see advanced development work start in the second quarter of this decade. Within the next 18 months sufficient ground work should be laid that a detailed 10-year plan covering these forms of advanced propulsion can be generated. There are even some areas where work involving the period 1970 to 1980 should begin.

C. Summary and The Future

J. I. Shafer

The three months' effort by the study group has been primarily one of familiarization and orientation. During the next three months, a more critical look will be taken at systems and component problem areas. It is hoped that it will culminate in a documentation of the entire study in the form of an external report; if Lab commitments on other programs permit further work, a preliminary design for our first electric-powered spacecraft would then be initiated on a minimum effort basis.

A portion of the study has progressed more rapidly than originally expected. Industrial and NASA work on thrust units, conversion systems, and reactors was well under way, and companies were very cooperative in discussing their programs with JPL personnel.

The Committee's studies have concentrated strongly on the *Snap-VIII*-ion propulsion system. One should not interpret this pronounced emphasis throughout the report to mean that JPL has examined and now endorses this system however; at this stage the committee's position is one of evaluating the *Snap-VIII* system and its applicability for deep space missions. A high priority, and emphasis, exists because Aerojet-General and others are currently developing such a system under NASA contract and guidance.

The Committee has arrived at some preliminary conclusions; based on system considerations, it notes some problem areas that warrant early attention. These are summarized as follows:

(1) It is urged that the ion engine flight experiment to evaluate beam neutralization be initiated as soon as possible, preferably by 1962. (Latest NASA plans indicate 1964.) Ground testing and the national effort on advanced propulsion are increasing rapidly. The flight experiment is crucial in determining whether these efforts and associated costs are justified.

(2) There is no apparent need for a 30 Kw propulsion system for interplanetary missions. In order to be useful it must be 60 Kw, or possibly larger, at the specific weight of 50 lb/Kw.

(3) At this early stage in the spacecraft system studies, payload capabilities for deep space missions with a *Centaur–Snap-VIII* vehicle are very sensitive to numerous basic assumptions and are fluctuating appreciably, from significant payloads to zero or less. Propulsion contractors should be urged to reduce the specific weight of the propulsion system to less than 50 lb/Kw at every opportunity. JPL should try to refine their payload estimates.

(4) There may be a serious problem concerning spacecraft control because of the abnormally low natural frequency of the Aerojet-General radiator design. A more thorough examination is urgently needed by JPL guidance and control personnel. This problem may have serious enough consequences as to require a complete change in spacecraft configuration.

(5) It is very desirable to defer a choice of the type meteoroid protection one uses for the radiator as long as is practical in order to capitalize on current efforts to improve bumper design criteria. Note that adoption of the conservative "armoured" design (currently under consideration by NASA) would increase the radiator weight enough over the "bumper" design to wipe out the net payload for all valuable deep space missions according to JPL's current estimates.

(6) From a systems viewpoint, i.e., because of spacecraft packaging advantages, it is not necessarily desirable to design the radiator for a minimum weight. In addition, control requirements and spacecraft weight may be reduced somewhat for radiators of smaller area, even though the radiator itself is heavier.

(7) From the system viewpoint the Committee believes it is extremely desirable to test the power conversion system in the radiation environment of the reactor during ground tests. It is our understanding, however, that such a requirement has not been placed on the Aerojet-AI acceptance test for the reactor conversion system.

(8) If spalling material inside the radiator tubes (from meteoroid bombardment) will impair operation of the system, because of bearing contamination, etc., then spalling will be a more important design consideration than tube penetration. This requirement alters the design and/or increases the weight.

(9) It is our understanding that no responsibility for the AC-to-high-voltage-DC conversion system has been assigned as yet. Because system factors influence its overall design, and some development effort appears necessary, consideration should be given to further study and assignment of responsibility to assure spacecraft compatibility and component availability on schedule.

(10) Operation of the complete propulsion system at a reduced power level (10% of the nominal), as specified in the contracts, may be very desirable for some *Snap-VIII* deep-space missions in order to extend its lifetime.

Items 1, 3, and 4 are especially important problems and warrant prompt action by JPL.

Although the preceding sections of this Status Report have tended to highlight the problems and difficulties with ion propulsion and electric spacecraft, it is important that the reader be given a proper perspective as to electric propulsion utilization. In general, the group in its studies to date has found nothing but reasons for optimism about the engineering feasibility and state of the art. A deeper examination of the problems is necessary, however.

The question of technical feasibility remains unanswered today, but here again, key individuals appear rather optimistic. The ion beam neutralization analysis discussed by Dr. Meghreblian should be well along in about two months and should be a major input in guiding the ion-engine flight experiment and enhancing our fundamental understanding of the phenomenon.

Incidentally, it should be noted that the Laboratory's Advanced Propulsion section will be following technical progress of alternate propulsion systems having performance that is essentially competitive with ion propulsion. There is always a possibility that colloid propulsion and MHD propulsion, which have less severe or no beam neutralization problem, would develop faster than expected. Alternately, an ion system which emits both positive and negative ions could assume importance if a good, simple source of negative ions were to be developed; here again, beam neutralization is much simpler. These possibilities merely indicate that the advanced propulsion field is in a rapid state of flux and warrants close observation.

The Section on Reactor and Shielding touched on the very sensitive political problem of atmospheric contamination by flying nuclear reactors. Here again the outlook appears somewhat brighter than several months ago as a result of the recommendation by the Aerospace Nuclear Safety Board that all *Snap* systems be given approval for flight if certain practical criteria are met. There is no final assurance today that this hurdle has been cleared; however, the picture appears much brighter. Note too, that for several months NASA Headquarters has been pressing for a resolution to this question.

Finally, is should be mentioned that the study has concentrated on the *Atlas-Centaur* and a power-limited propulsion spacecraft. Consequently, the performance potential of electric systems has not been covered.

Future propulsion developments and larger spacecraft will produce significant gains in reliability and performance. One can expect the specific weight of electric propulsion systems to drop from its present 50 to 65 lb/Kw to about 5 to 10 lb/Kw; dynamic power conversion systems, such as today's turbo-alternator, might well be supplanted by one of the simpler static systems, such as the plasma diode.

In reality, it is with the larger *Saturn*-size vehicle that electric propulsion comes into its own and reveals that the electric-powered *Atlas-Centaur* is an important stepping stone toward truly high-performance vehicles. As an example, estimates of the payloads for a Mars circular orbit with a *Saturn C-2*, with and without a 1 Mw ion system, indicate the ion spacecraft would have about six times the gross payload capability of the chemical spacecraft—about 19,000 lb vs 3200 lb respectively for 230-day transit times.

It is performance potential of this type and provocative problems of the nature indicated in earlier presentations that has always stimulated the Laboratory's imagination in the past. Possibly once again, with the incentive of electric propulsion performance and the challenge of its problems, the Laboratory has an opportunity to demonstrate its initiative and ingenuity by developing and utilizing this type propulsion for spacecraft.

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